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Contract No. Nonr-1675(00)

DUCTED PROPELLER ASSAULT TRANSPORT

Summary Report

Report No. D181-945-001

15 May 1956

BELL *Aircraft* CORP.

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DUCTED PROPELLER ASSAULT TRANSPORT

ONR **ARMY**

A Combined
Research and Development Program

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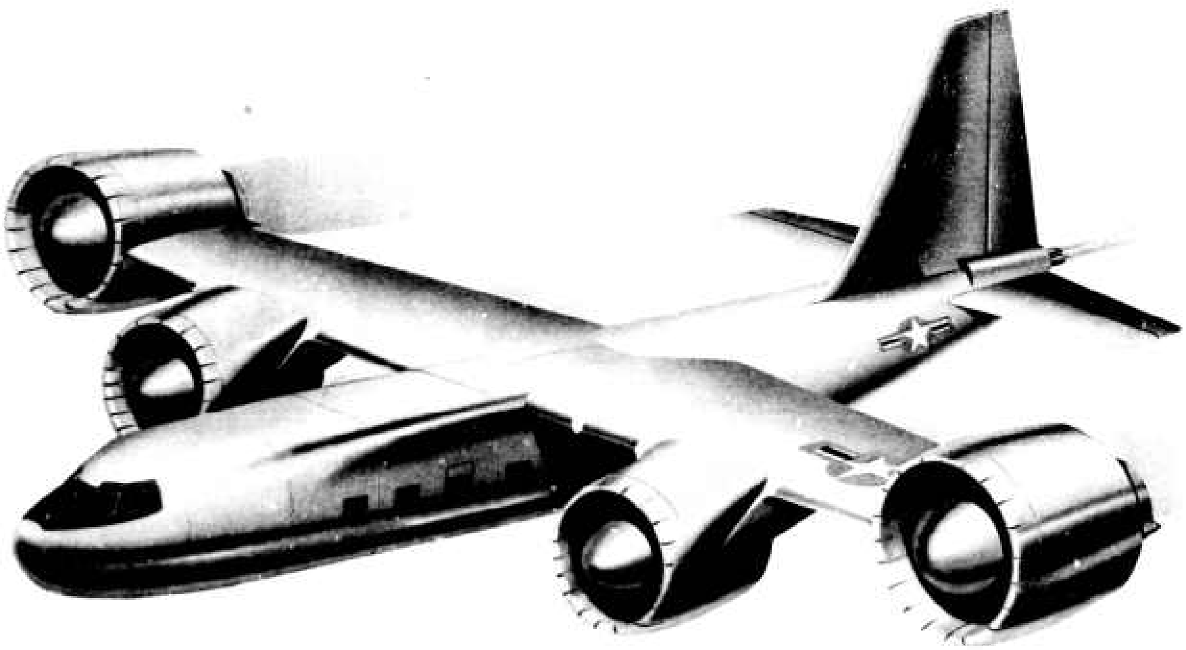

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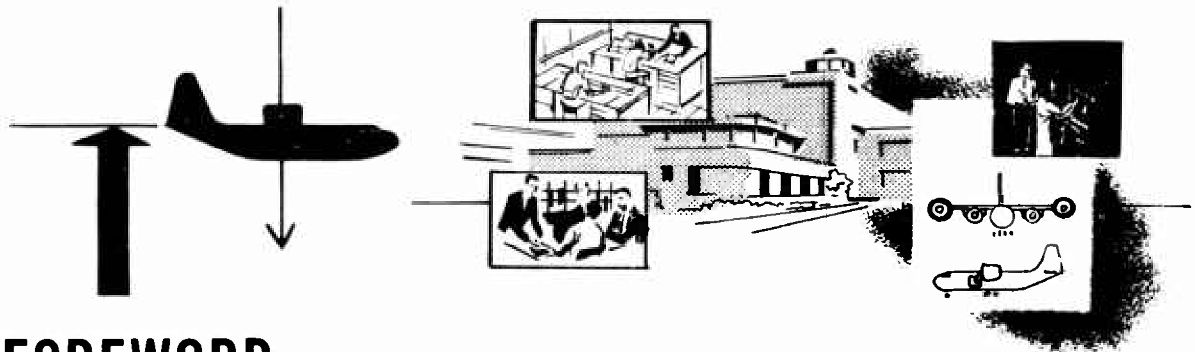

FOREWORD

Contract Nonr-1675(00) was awarded to the Bell Aircraft Corporation by the Office of Naval Research under sponsorship of the Army Transporation Corps. This is one of a series of five study contracts let to investigate the application of various schemes to the design of Vertical Take-off and Landing (VTOL) or Short Take-off (STO) Assault Transport Aircraft. The study program was conducted during a twelve-month period ending on 15 May 1956.

The particular field of investigation at Bell Aircraft is the application of ducted propeller propulsion systems to the design of aircraft capable of performing the Assault Transport mission. The results of the investigation are presented in the following reports:

Title	Report Number
Summary Report	D181-945-001
Design Report	D181-945-002
Survey of the State of The Art	D181-945-003
Performance	D181-945-004
Stability and Control	D181-945-005
Duct and Propeller Analysis	D181-945-006
Preliminary Structural Analysis	D181-945-007
Standard Aircraft Characteristics	D181-945-008

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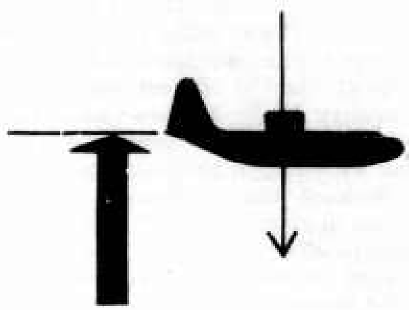

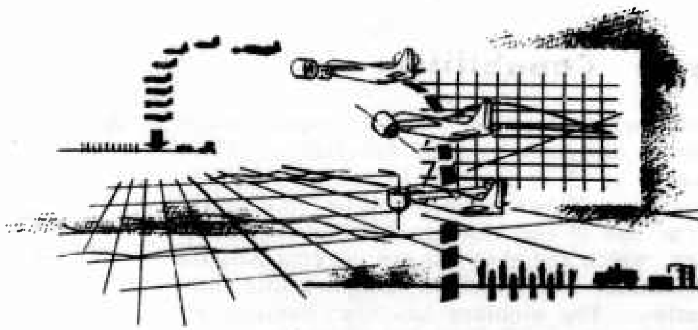

SUMMARY

Requirements

During the period from 1 May 1955 through 15 May 1956, Bell Aircraft Corporation has conducted the Ducted Propeller Assault Transport Study for the Air Branch, Office of Naval Research and the Army Transportation Corps. At the initiation of the study the following listed aircraft requirements were selected for the design:

Payload	Outbound - 8000 pounds	Inbound - 4000 pounds
Take-off	Vertical	
Cabin Section	8 feet x 9 feet	
Cargo	35 infantrymen or equivalent wheeled vehicles	
Ambient Take-off Conditions	95° F at 6000 feet Altitude	
V Cruise	300 mph	
20% of Radius at Sea Level to Target and Back		
Landing	Vertical	
Radius	425 statute miles	

To remain controllable under one engine out conditions and be able to make a "controlled crash" landing.

During the course of the study, a good portion of time was used to evaluate and study the physical principles behind the application of ducted propellers to VTOL aircraft. The practical studies of applying this knowledge to airframe design was performed with the result that an assault transport configuration has been selected which will very adequately perform the required mission.

System Capabilities

The ducted propeller VTOL transport which has been designed by Bell Aircraft is shown in Figure 1. It is powered by six Allison 550-B1 engines. These engines will become available for use in the 1959 period and the power ratings, which are shown in the current Allison specification, were used in designing this configuration. The airplane has been designed to accomplish the basic mission which has been covered in detail by a succeeding section. A gross weight of 67,380 pounds is required to cruise the 425 miles radius. The airplane has a 98-foot wing span, over-all length of 81 feet and an over-all height of 34 feet. A four-duct propulsion system was chosen for this application, involving the use of an 11-1/2-foot contra-rotating propeller in each outboard duct, and a single rotation 8-1/2-foot propeller in each inboard duct. The airplane has been designed to carry 35 airborne troops or an 8000-pound payload during the vertical take-off and landing condition at 6000 feet altitude on a 95°F hot day. The vertical take-off is accomplished by swiveling the ducted propellers into a vertical position, and expanding the flaps on the inlet of the shroud to develop a high static thrust during the take-off procedure. The ducts rotate into a level flight position as the airplane transitions into level flight. In order to accomplish this take-off at 6000 feet on a 95° day, it has been assumed that water injection will be applied to the Allison 550 engine.

The configuration has the performance which is shown in Figure 2. The objective at the start of the study was a minimum cruise speed of 300 miles per hour. Performance analyses indicate that this airplane is able to exceed 500 miles per hour over a large altitude range for most of the gross weights involved and loading conditions visualized. The rate of climb at sea level is a maximum of 13,000 feet per minute and falls to 9000 feet per minute for the full 70,000 pounds gross weight. The high speed capability is of great importance in terms of combat zone evasion, greatly reducing exposure time when travelling in or near enemy held territory. The take-off flight paths which can be used by the transport are demonstrated in Figure 3. A vertical rise to 50 feet and a level transition to forward flight has been assumed. Two conditions of take-off are shown; one for no wind, and one assuming a practical operation condition of a 40-knot wind condition. For the no-wind condition, approximately 15,040 feet of air space is used to effect transition over a 27-second time period, moving from vertical take-off to level flight speed. A 40-knot wind, which may be encountered in operation in the field, helps in the take-off by reducing the transition distance to 360 feet and the total time for take-off to 19 seconds. The landing procedure is effected by rotating the ducts somewhat forward of vertical to decelerate over a 29-second time period, at which point a vertical descent is made, completing the landing cycle in 42 seconds. It can be noted here that the time periods for the take-off and landing procedures are very short in comparison to the total time of flight. Since it is short, the probability of engine failure is very small. It is very desirable to obtain maximum safety for engine failure during the hovering and transition phases of flight, but unduly compromising the aircraft design for this very small part of the total flight time may be somewhat unrealistic.

The STO/VTO capability of the horizontal attitude VTOL transport design is demonstrated in Figure 4. A vertical take-off can be performed at 72,000 pounds gross weight, where the thrust-to-weight ratio is 1.0 at the high altitude and temperature condition. Overload conditions of up to 18,000 pounds can be obtained by using short take-off ground runs up to 900 feet. If the take-off criterion is distance over a 50-foot obstacle, the additional ground distance consumed is indicated. For example, the gross weight of the aircraft capable of take-off over a 50-foot obstacle within a 500-foot distance is seen to be only slightly greater than the pure VTOL maximum gross weight. This implies that the gains to be made by using short take-off configurations appear rather limited and that combination VTO/STO aircraft appear quite attractive.

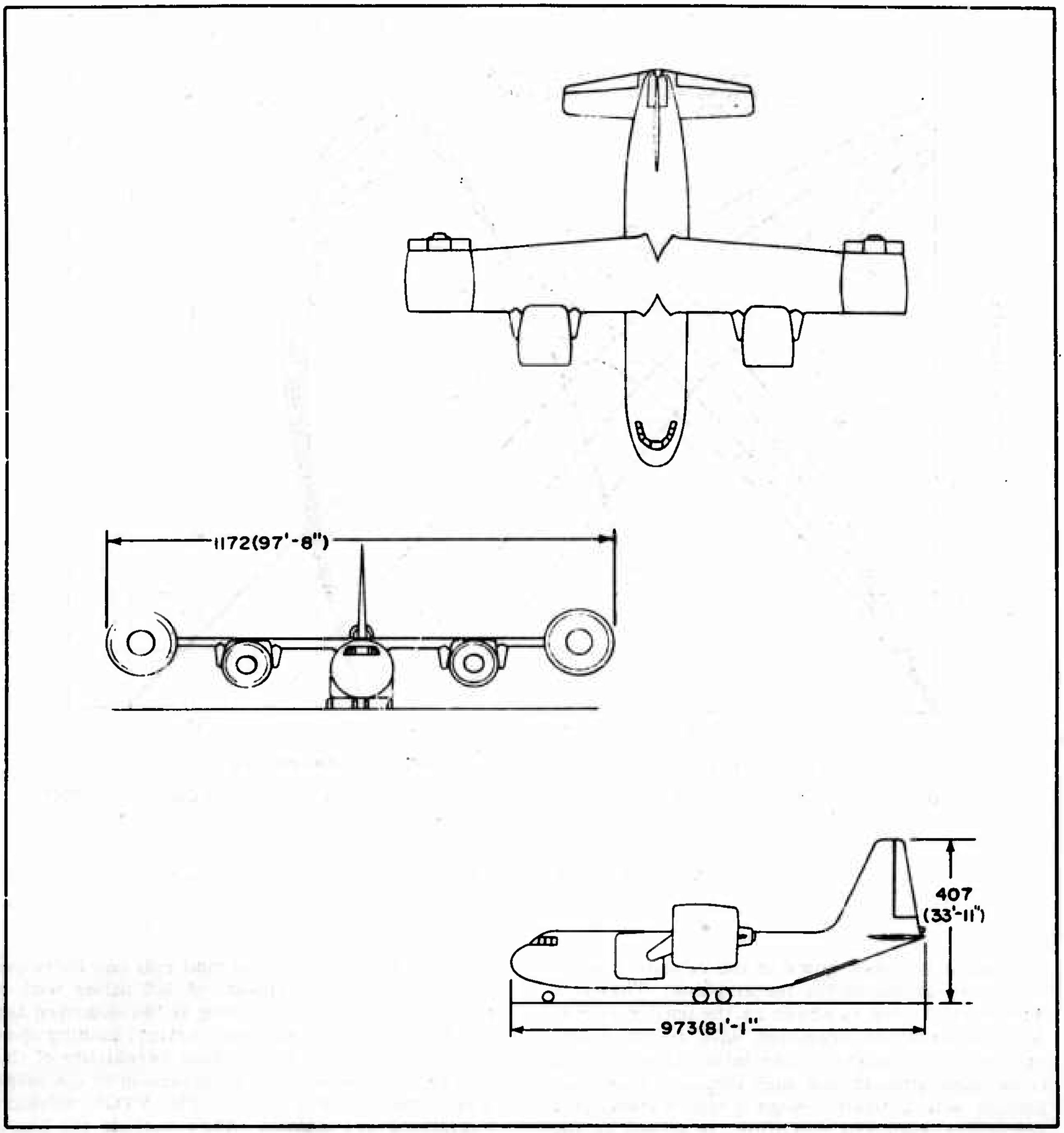

Figure 1. Assault Transport Basic Configuration

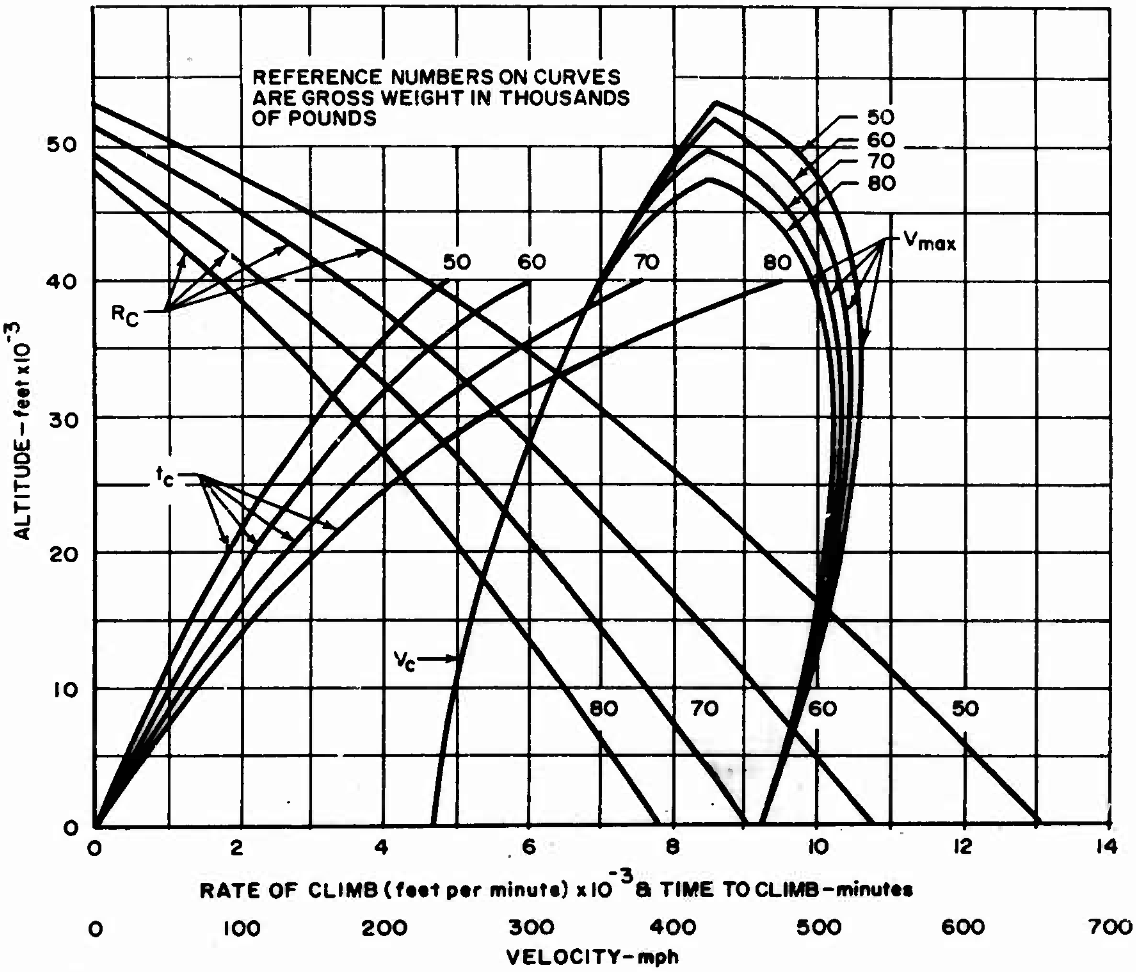

Figure 2. Classical Performance

Figure 5 shows some of the missions which have been examined for the airplane. The basic 425-mile mission is shown as the bottom curve. All the missions examined have included 20% operation at sea level. The influence of moving from high altitude and high temperatures conditions with a basic design gross weight, to a sea level standard condition, is shown in the second profile. An increase in the basic radius to 705 miles is obtained for this condition. A 770-foot short take-off ground roll can increase the radius to a maximum of 987 miles with a vertical take-off and landing at the outboard leg of the mission and another vertical landing upon return to base. The extreme versatility of the aircraft can be seen by examination of the mission chart in Figure 6. The VTOL mission variations are shown. They include the basic mission, a low-level basic mission at 11,300 feet cruise altitude, a high-speed mission with

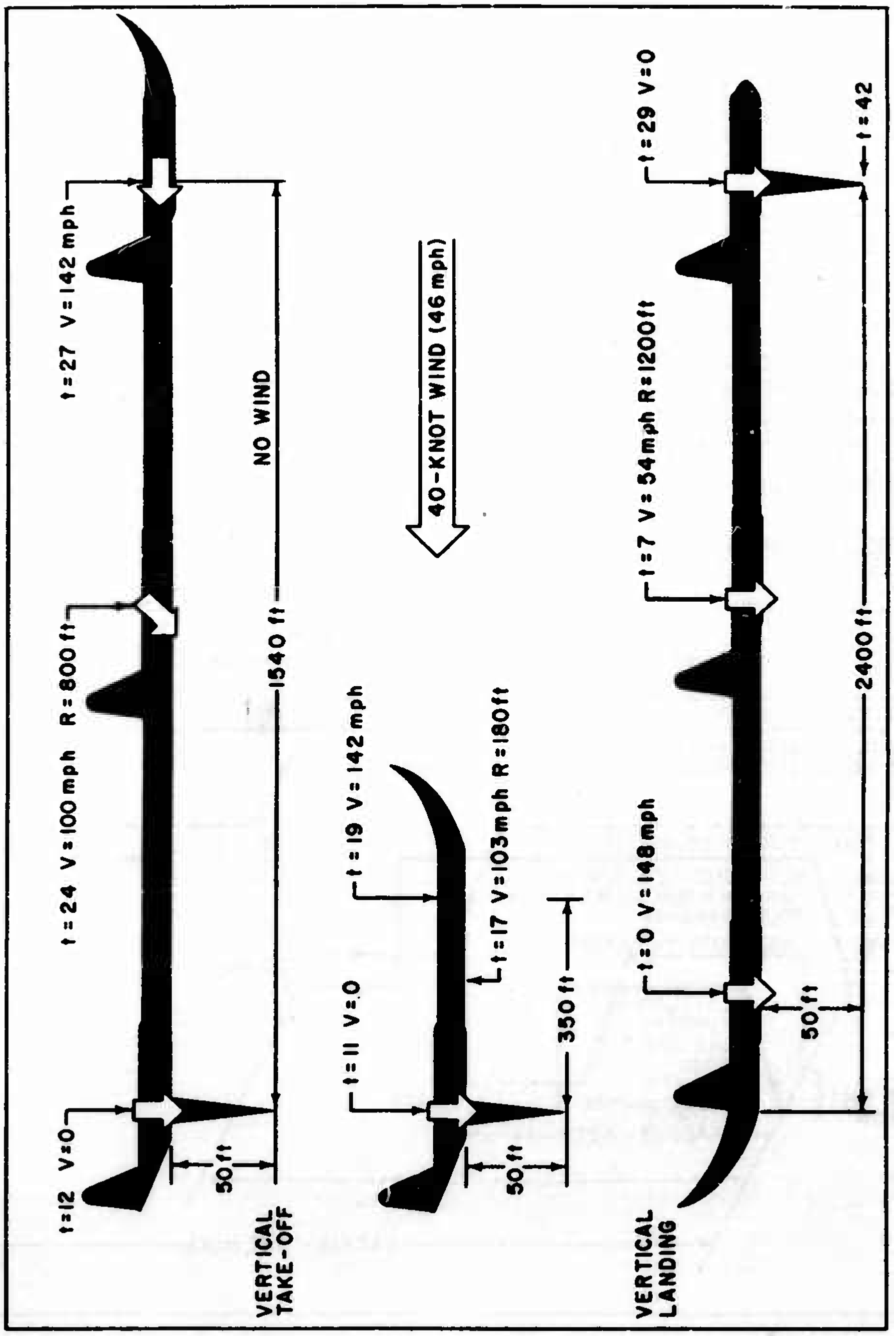

Figure 3. Take-Off Flight Paths

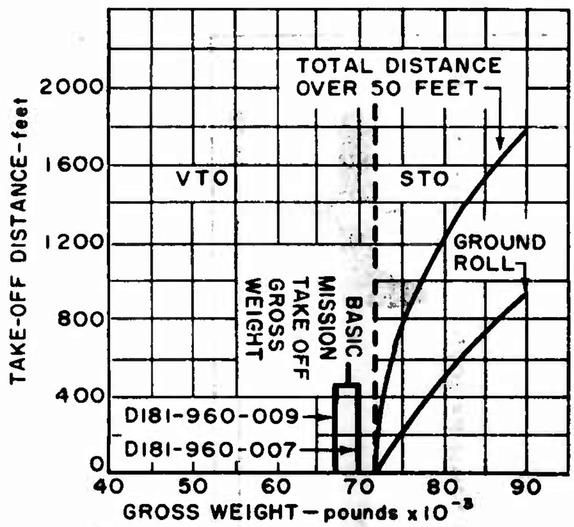

Figure 4. Take-Off Distance

cruise at 420 miles per hour, and a maximum speed mission at 455 miles per hour. The STO missions include the maximum radius mission of 987 miles, a maximum payload mission with a 16,700-pound cargo, and a high-speed mission at 450 miles per hour cruise speed. The aircraft has a maximum ferry range of approximately 3200 miles with a 770-foot ground run STO.

The hovering capabilities of the aircraft are quite good when it is considered that this is not the primary design condition. As seen in Figure 7, a hovering time of seven minutes is available at the extreme VTO range. As the range is decreased, the hovering time can be increased to a maximum of 74 minutes, including the altitude hot day condition. For sea level standard conditions, hovering time can be increased to 100 minutes by the allowable addition of fuel. The hovering times in conjunction with STO missions are also included.

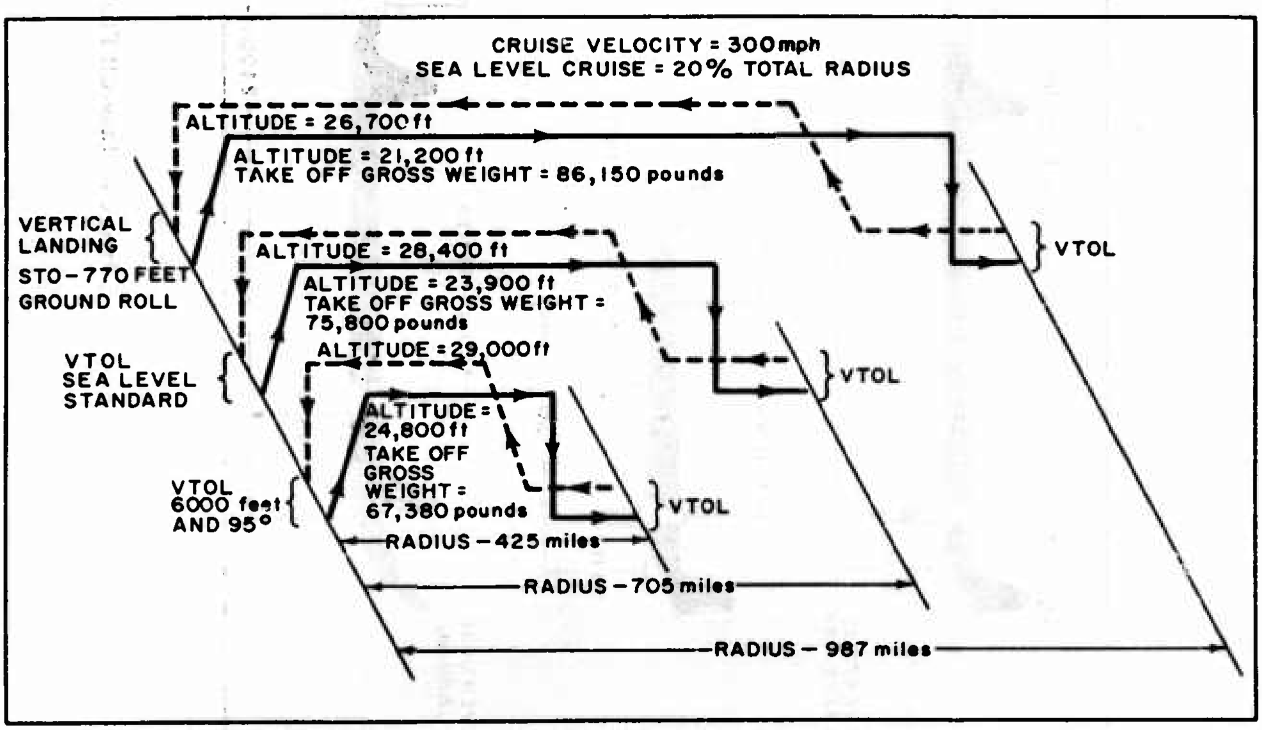

Figure 5. Mission Profiles

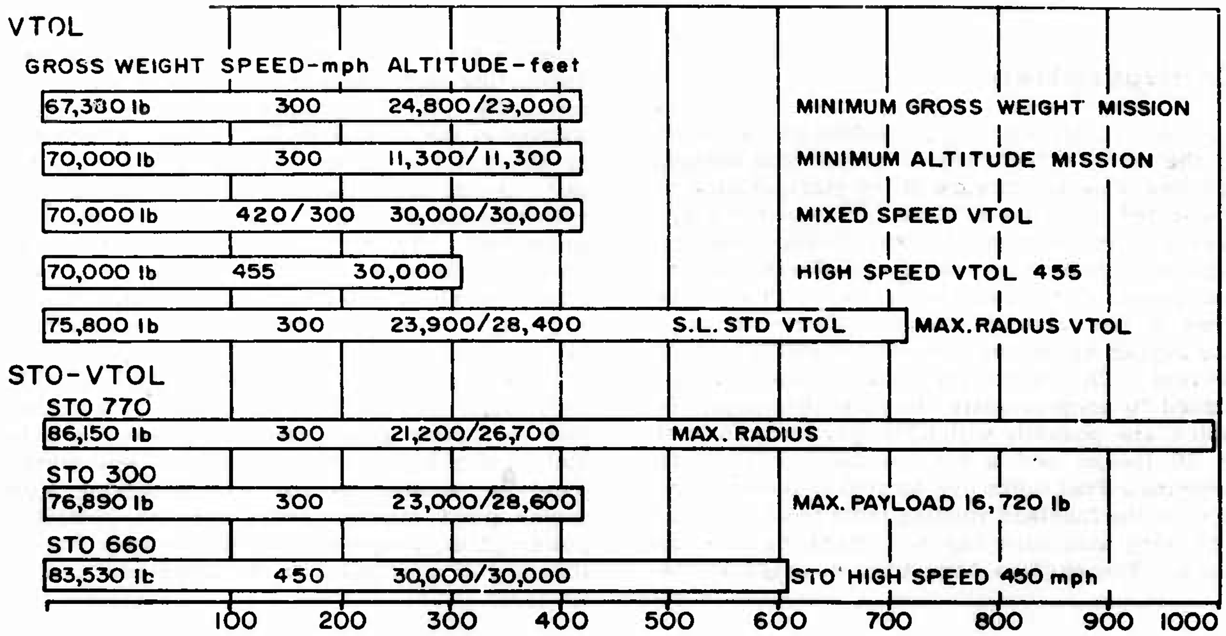

Figure 6. Mission Variations

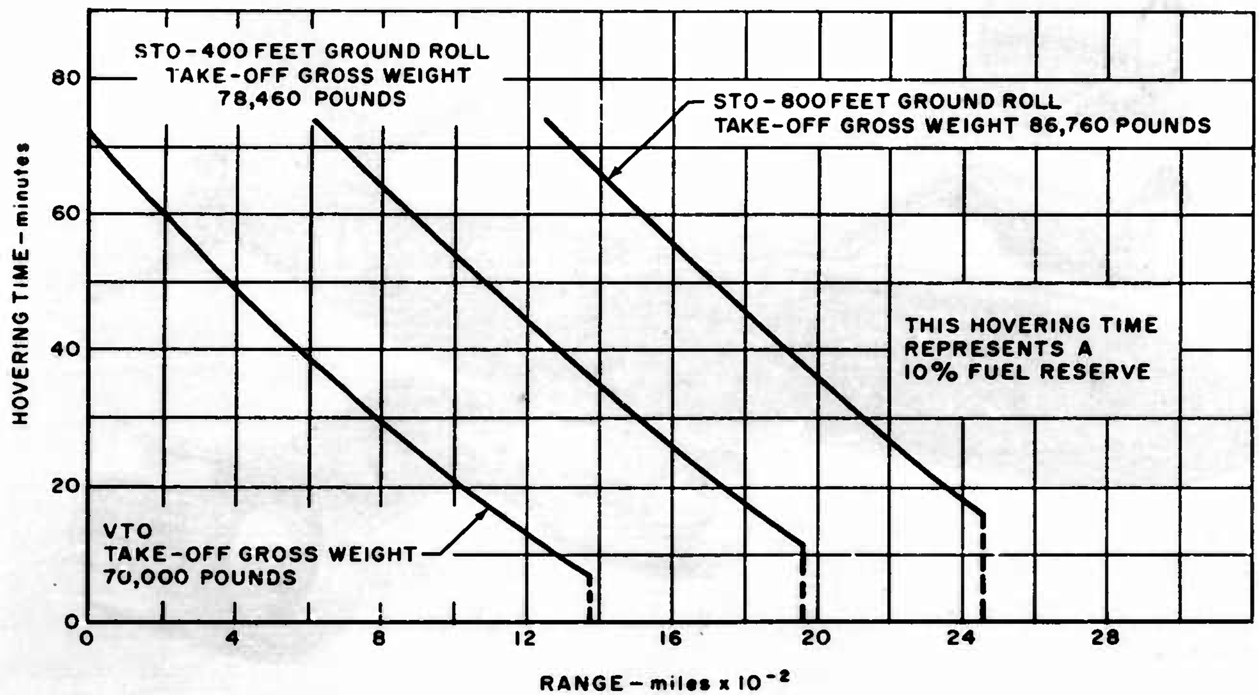

Figure 7. Range and Hovering Time

Configuration

The aircraft which is presented as the result of the study is basically a conventional design. A cutaway sectional view of the configuration is presented in Figure 8. It is evident that the wing, fuselage, tail, and landing gear construction are typical of present-day fabrication methods. Examination of the fuselage shows a three-man crew in the forward section. Electronic gear and oxygen equipment are located just aft of this section. The cargo compartment has been designed to accommodate the increased payloads which are possible with STO operation. A load of 30 troops and a 4 x 4 weapons carrier is depicted. Fuel tanks are located in the top portion of the fuselage running from front to rear. The wing structure has no cutouts for flaps or slats. The engines have been mounted in the centerbodies of the ducts so there is no interconnecting shaft running through the wing structure. The reaction control source, a J85 engine located at the aft end of the fuselage, is mounted at the junction of the vertical and horizontal tail. Aside from the ducts themselves, a very conventional aircraft construction problem is presented. There are no large surfaces to move and there are no large cutouts in the wing structure with complicated mechanisms. There are no secondary systems such as boundary layer control. It is felt that this particular transport configuration could be built quickly and easily by conventional methods, and that this particular arrangement of power plants is quite suitable for conversion from one power plant to another or for subcontracting with power plant manufacturers for the complete power plant, propeller, and duct assembly up to the point of attachment on the airplane.

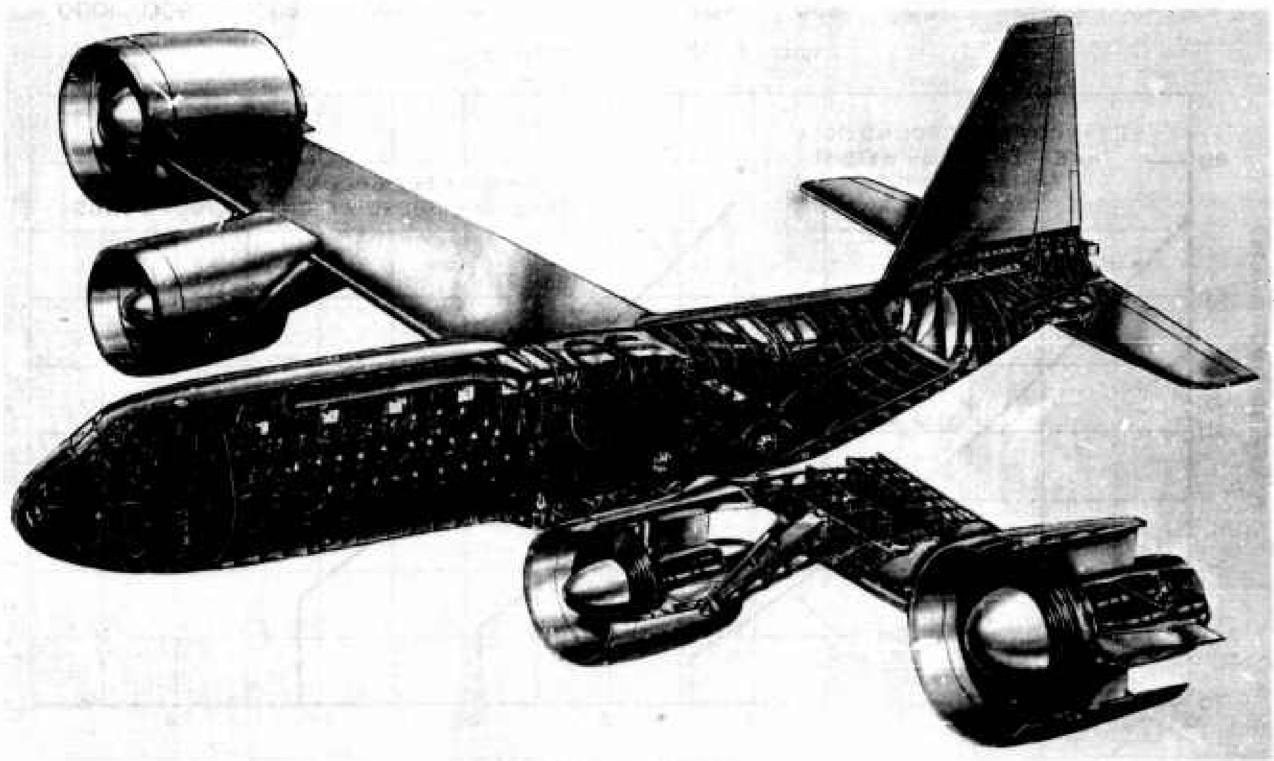

Figure 8. Cutaway of Assault Transport

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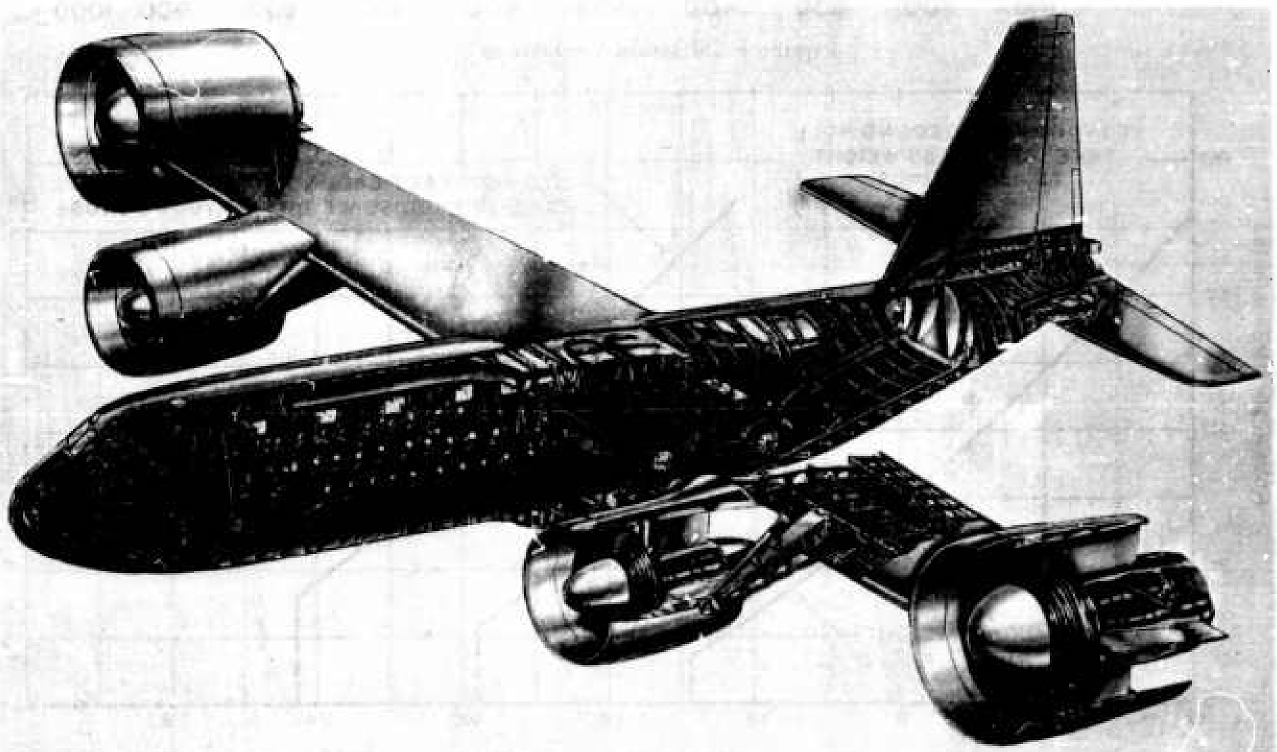


Figure 8. Cutaway of Assault Transport

A weight summary of the airplane is presented in Table I. The basic mission gross weight is 67,380 pounds. The empty gross weight is about 43,000 pounds. It is thought that this is a reliable weight estimate, and that the airplane could be built for approximately this quantity. Some of the payloads that are possible with the airplane are shown in Table II. VTOL missions and STO loadings are presented. The compartment capacity permits the loading of 59 airborne troops or 52 litter patients. Three rows of troop seats can be installed to seat the troops or three bays of litters will accommodate the litter cases. Other vehicular payload capabilities are listed. The rear loading ramp and doors provide easy access to the cargo compartment and facilitate quick unloading.

TABLE I. ASSAULT TRANSPORT WEIGHT SUMMARY

Item	Weight
Wing Group	5200 pounds
Tail Group	1171 pounds
Body Group	7423 pounds
Landing Gear Group	2230 pounds
Ducts — Inboard	2780 pounds
Outboard	3920 pounds
Propulsion	
Engines (6 Allison 550-B1)	9450 pounds
Inboard Gear Boxes (2)	980 pounds
Outboard Gear Boxes (2)	2180 pounds
Mounts and Supports	1330 pounds
Engine Systems	1055 pounds
Propellers — Inboard	1094 pounds
Outboard	1522 pounds
Fixed Equipment	3480 pounds
Total Weight Empty	43,815 pounds
Crew	645 pounds
Fuel, oil and water	14,920 pounds
Payload	8000 pounds
Total Useful Load	23,565 pounds
TOTAL GROSS WEIGHT	67,380 pounds

TABLE II. PAYLOAD CAPABILITY

	VTOL Missions	
35	Airborne Troops	8000 pounds
38	Litter Cases	8000 pounds
1	Field Ambulance	5485 pounds
3	Jeeps	7350 pounds
2	Loaders, shovel 1/3 cubic yard	8600 pounds
1	Truck 1/2 ton	4775 pounds
1	Field Ambulance 3/4 ton	6880 pounds
	STO Missions	
59	Airborne Troops	13,700 pounds
52	Litter Cases	11,000 pounds
2	Field Ambulances, 3/4 ton	13,760 pounds
2	Cargo Carrier M29C	11,950 pounds
1	Scout Car M3A1	13,055 pounds
1	Tractor, High Speed, M2	14,915 pounds
2	Trucks, 1/2 ton	9,550 pounds
1	105 mm Howitzer and 3/4 ton Weapons Carrier	11,000 pounds

Propeller and Duct Design

In addition to this airplane's capabilities, a considerable amount of time has been devoted to analysis of the ducted propeller units. At least three different types have been found to be suitable. The first two types appear in Figure 9. On the design which is presented, there are two variable-pitch 12-bladed contrarotating propellers in the outboard ducts. These propellers do not require any inlet vanes or exit stators to straighten the flow. The design has been checked over the speed range concerned for

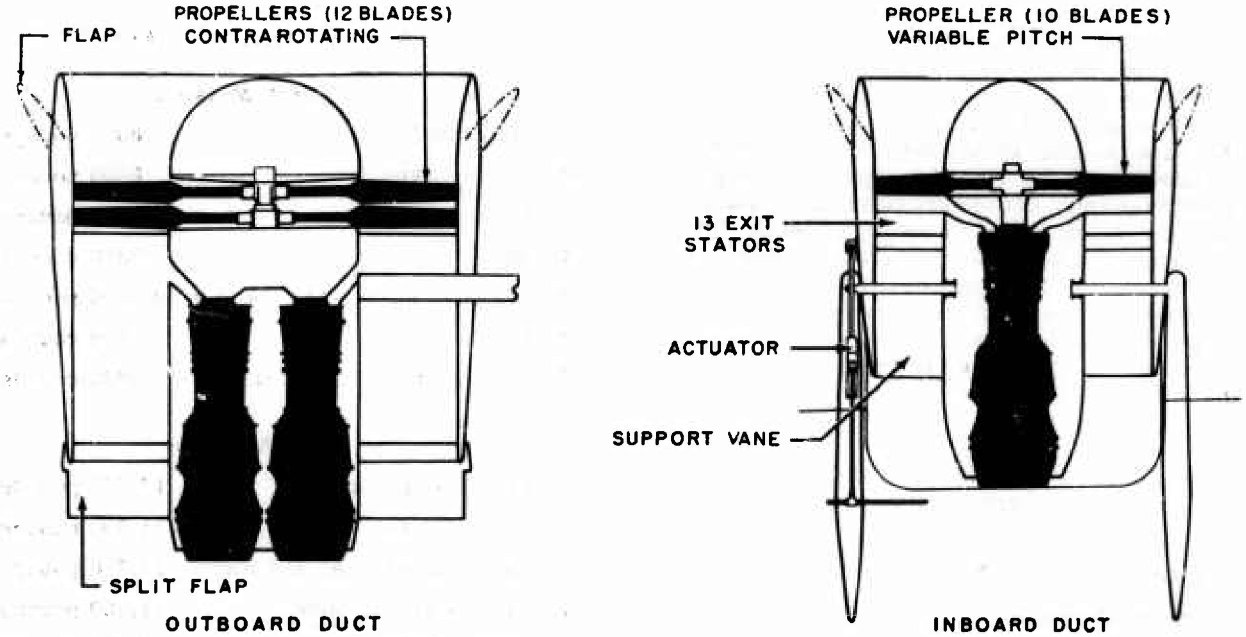


Figure 9. Duct and Propeller Cutaway

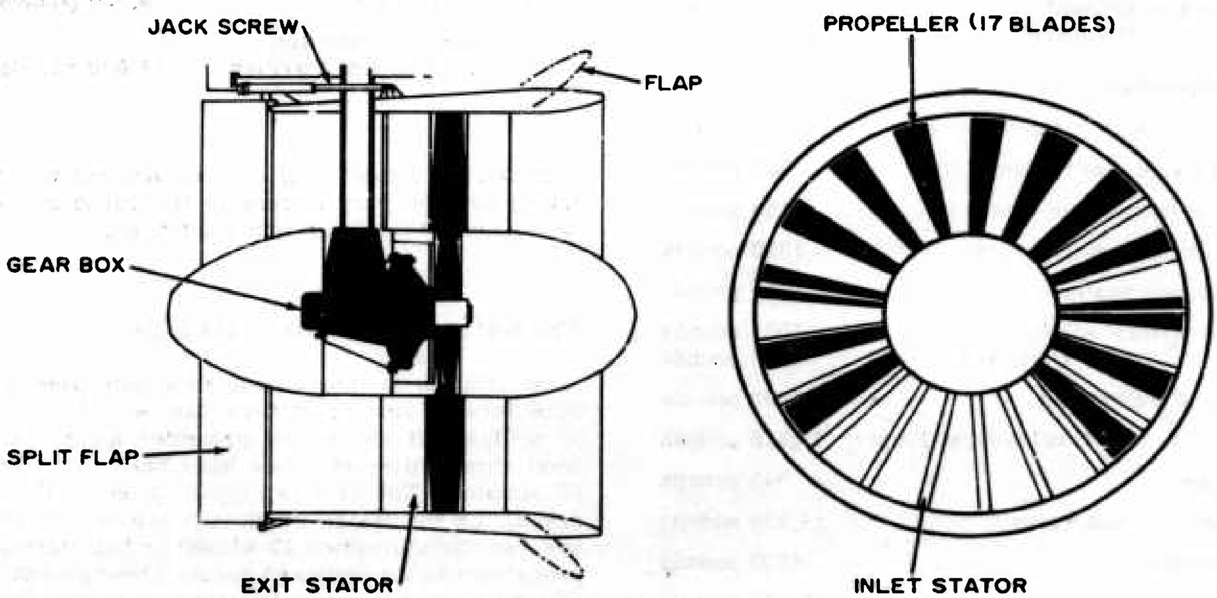


Figure 10. Fixed Rotor Ducted Propeller

off-design and partial power considerations and is a very workable solution. A single variable-pitch propeller has been used for the inboard duct, and a fixed pitch exit stator is used to provide flow straightening as the flow comes out of the duct itself. In this case, a single engine has been mounted in the hub of the propeller. The third solution is represented by Figure 10 where a multibladed fixed rotor is used in conjunction with a set of variable inlet vanes and fixed exit stators. This does not require any variable mechanism on the rotating parts of the propeller and it can be made to operate effectively by varying the geometry of the inlet guide vanes. Each one of these three solutions has been checked against the flight speed range of operation, and any one of them is a good solution to the ducted propeller problem. At the present time, a choice has not been made as to the approach that should be followed. It is thought that this is a case where future mechanical studies should be linked with propeller manufacturing requirements.

Estimates of the drag of these shrouded propellers have been incorporated into the performance analyses. Throughout the study, considerable comments have been made on the effect of duct drag, and it is agreed that an improperly designed shroud will result in a very high drag increment. Drag increments as high as the total airplane itself are possible. Figure 11 shows the shroud drag at sea level and at 40,000 feet is approximately one sixth the drag of the total airplane. This figure is a conservative estimate of shroud drag in that any alleviating effect on induced drag has not been accounted for in the calculation. It is anticipated that the shrouded propellers will be rotated in such a direction as to alleviate induced drag, and a certain increase in aspect ratio is anticipated as a result of the large shroud. These alleviating factors have not been incorporated in this drag estimate. At the same time, this level of drag is roughly comparable to the type of drag level which would be encountered with certain strut-mounted turbojet engines. In other words, a conservative estimate of shroud drag has been considered. The airplane performance is still very attractive, indicating that shroud drag is not excessively high.

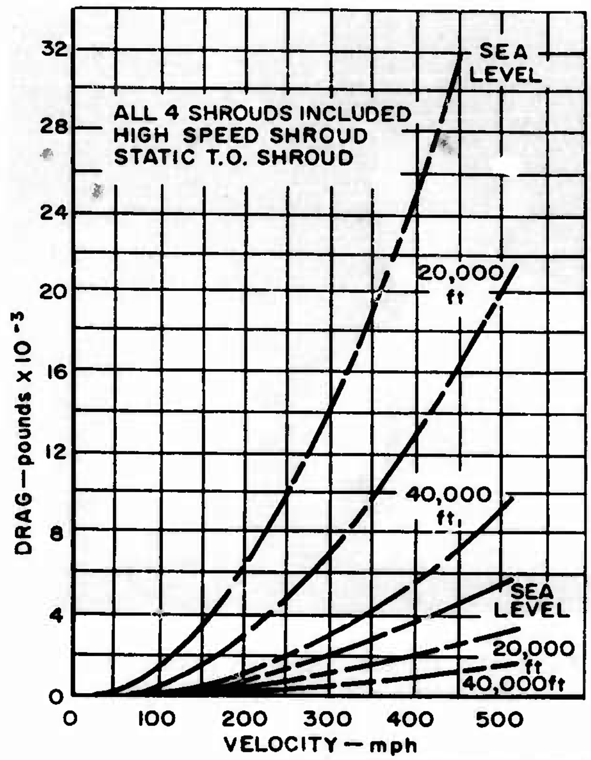


Figure 11. Shroud Drag

In order to verify the results of the analyses with some type of experimental information, a program has been initiated in conjunction with the University of Wichita to examine some of Bell Aircraft's fan designs. Figure 12 shows the multibladed rotor, the first model put into the Wichita tunnel, being balanced for mounting on the test stand. The next photograph (Figure 13) shows the partial assembly, and displays the 75 horsepower water cooled motor and the shaft arrangement which supports the rotor. The next picture, Figure 14 shows the total assembly mounted in the tunnel. The installation is shown with the high-speed shroud. Two configurations will be tested; one representing a static shroud with the inlet flaps extended, and the second a configuration which represents the inflight high-speed shroud condition. The shroud fineness ratio is very good, and the drag level is reason-

Figure 12. Ducted Propeller Tests. Propeller Balancing (University of Wichita)

able. The model will be used to establish transition effects on the shroud by rotating it through 90° at various velocity conditions to establish force and moment data and pressure distribution. The initial testing period has been completed, and preliminary indications are that the results agree quite well with the original analysis. The practical inlet flap configuration appears to be quite satisfactory.

Safety Considerations

Another problem of vertical take-off and landing airplanes which must be considered is the conditions which might exist in case of engine failure in hovering. There are two approaches to the failure problem. One approach involves just accepting the failure and dropping; the other is trying to fly with a reduced number of engines. In Figure 15 a rather conservative analysis is presented in that engine failure is assumed to occur instantaneously. It is assumed to occur at the hovering condition and all residual energy in the propeller disappears. Even under this conservative assumption it can be seen that

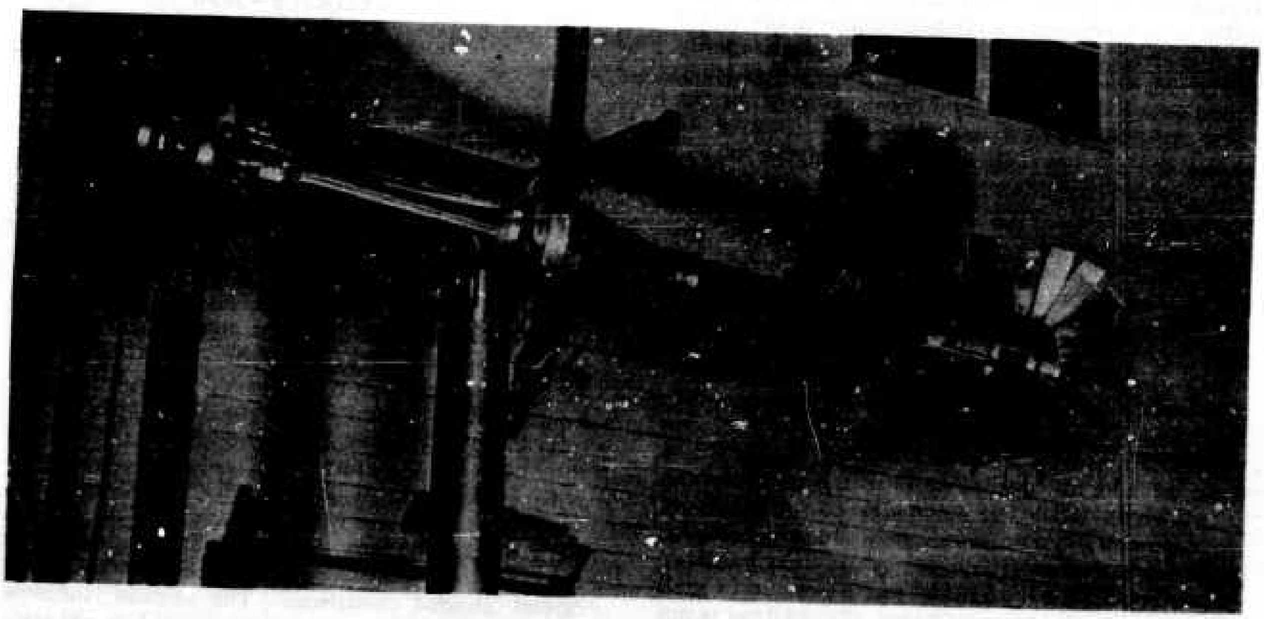


Figure 13. Ducted Propeller Tests. Partial Assembly at University of Wichita

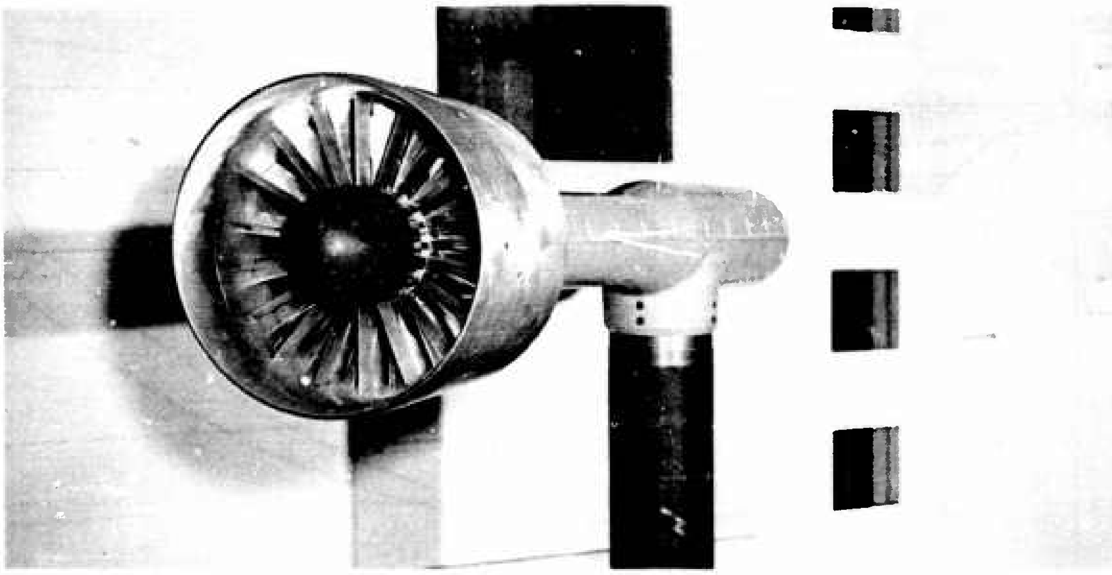


Figure 14. Assembled Model in Tunnel (University of Wichita)

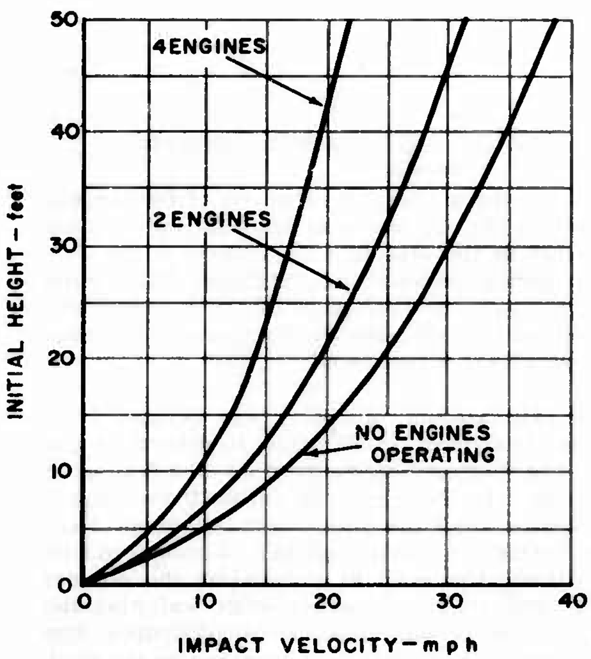


Figure 15. Engine Failure During Hovering

the impact velocity assuming 2 engines have failed, is only about 22 miles per hour from a 50-foot altitude. Successively lower heights of initial engine failure would give a reduced velocity effect. The double emergency situation up to the full engine failure condition has been examined. Even under those circumstances the impact velocity is 37-1/2 miles per hour. Velocities cf this order of magnitude wculd still allow the passengers and crew of this airplane good chances for survival. The airplane would probably be damaged, but the passengers would probably remain relatively uninjured.

The other case which was examined is shown in Figure 16. The shroud horsepower requirements throughout the transition region are shown as a ratio to the static horsepower requirement. It is presented in the same form as that presented by Mr. McKinney of the NACA in his IAS Preprint 597 when he compared the tilting wing airplane with the vectored stream airplane. First of all, the ducted propeller airplane followed basically the same horsepower requirements through the transition region as the tilting wing airplane. Neither one of these airplanes

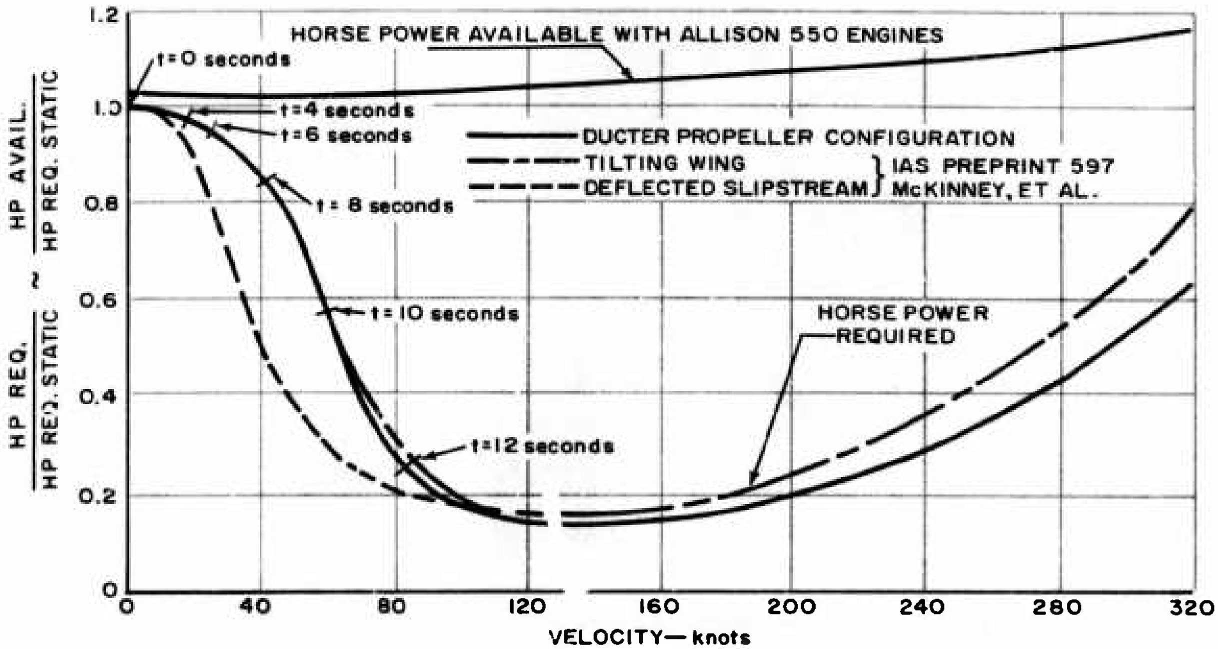


Figure 16. Transition Power Requirements

appears to have a reduced power effect comparable to the vectored slip-stream airplane. The second thing the curve shows, is the very short time period which exists from the hovering position to the level flight condition. Here a total of eight seconds will cover the time period during which a single engine failure would be apt to give trouble. In 12 seconds, the time period may have failures on one to four engines has been passed.

Once again, while ultimate safety is an important factor, the airplane design should not be overly compromised to give safety in this particular region of flight. The ducted propeller transport has been designed so that an automatic symmetrical shut down of engines will follow engine failure. Under those circumstances, control of the airplane is possible at all times, and as seen from this curve there is good possibility for operating at a reduced power condition.

Hovering Stability and Control

Bell Aircraft believes that one of the largest problems in any vertical take-off and landing aircraft is the stability and control of the aircraft during the hovering condition. This strong feeling comes as a result of the work on the jet-propelled test vehicles where this difficult problem was first encountered.

A fair amount of effort was devoted to it during this study in an effort to determine the stability control characteristics of a transport aircraft. In Figure 17 is shown the arrangement with which the pilot and the airplane characteristics are linked together. A cockpit mock-up allows the pilot to manipulate the normal stick and rudder controls while watching the airplane presentations on two oscilloscopes. The airplane representation is provided by the bank of REAC machines in the background. The

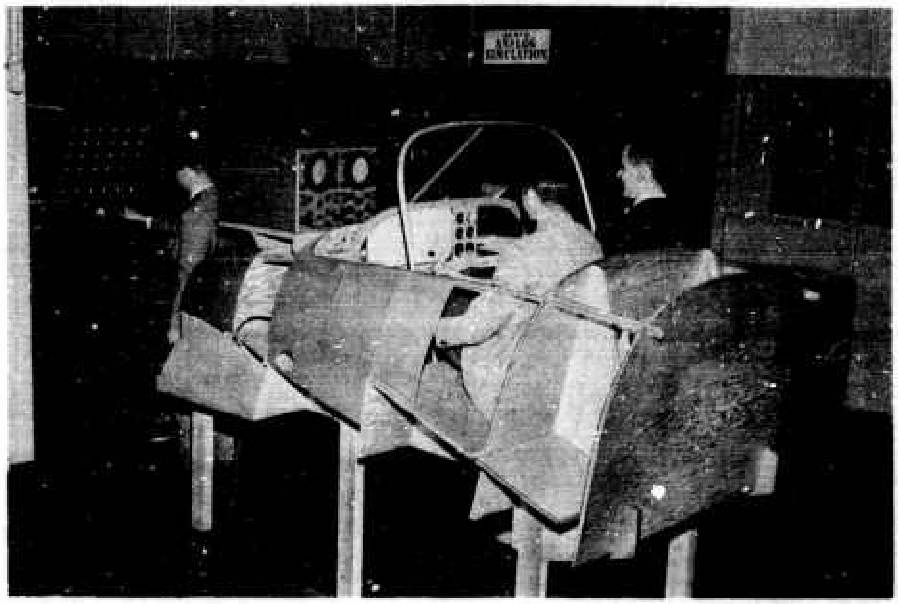


Figure 17. Computer and Cockpit Set-Up

longitudinal characteristics of the aircraft are set into one analog computer and the lateral characteristics into another computer. The computers are then coupled together, the pilot input fed into the circuits, and the resultant simulated aircraft traces are recorded. From these traces considered opinions can be formed on the problems of hovering stability and control.

Manual control gradient studies are shown in Figure 18 demonstrating the pilot's input as compared with the airplane's response. In this sample, three different control gradients were tried before the pilot agreed that there was satisfactory control of the airplane. This control gradient was then accepted as the requirement.

Other REAC records examining the influence of gusts in various planes are shown in Figures 50a and 50b. Gusty air is one of the practical problems which any hovering airplane will have to face. The REAC trace shows that the coupling moment in the ducted propeller transport is not as large as some of the turbojet propelled aircraft studies indicated. For example, a pitch gust has the major effect in the pitch plane, with small inputs of gust in other planes.

Some of the material required to put these problems on an analog computer is shown in Figure 19. Here is shown one of the generalized circuit diagrams which is used for hovering stability and control analyses. This diagram has been simplified by not closing the circuits. They are left open since the outputs are assumed to be inputs in the lower left hand side of the chart. The longitudinal computer is represented by the upper diagram and the lateral directional computer by the lower diagram. The dotted area indicates the interconnecting characteristics which have been used to get a reflection of lateral directional motion in the longitudinal plane. The circles represent the potentiometers by which the derivatives of the airplane are put into the circuit thereby actually representing the airplane itself.

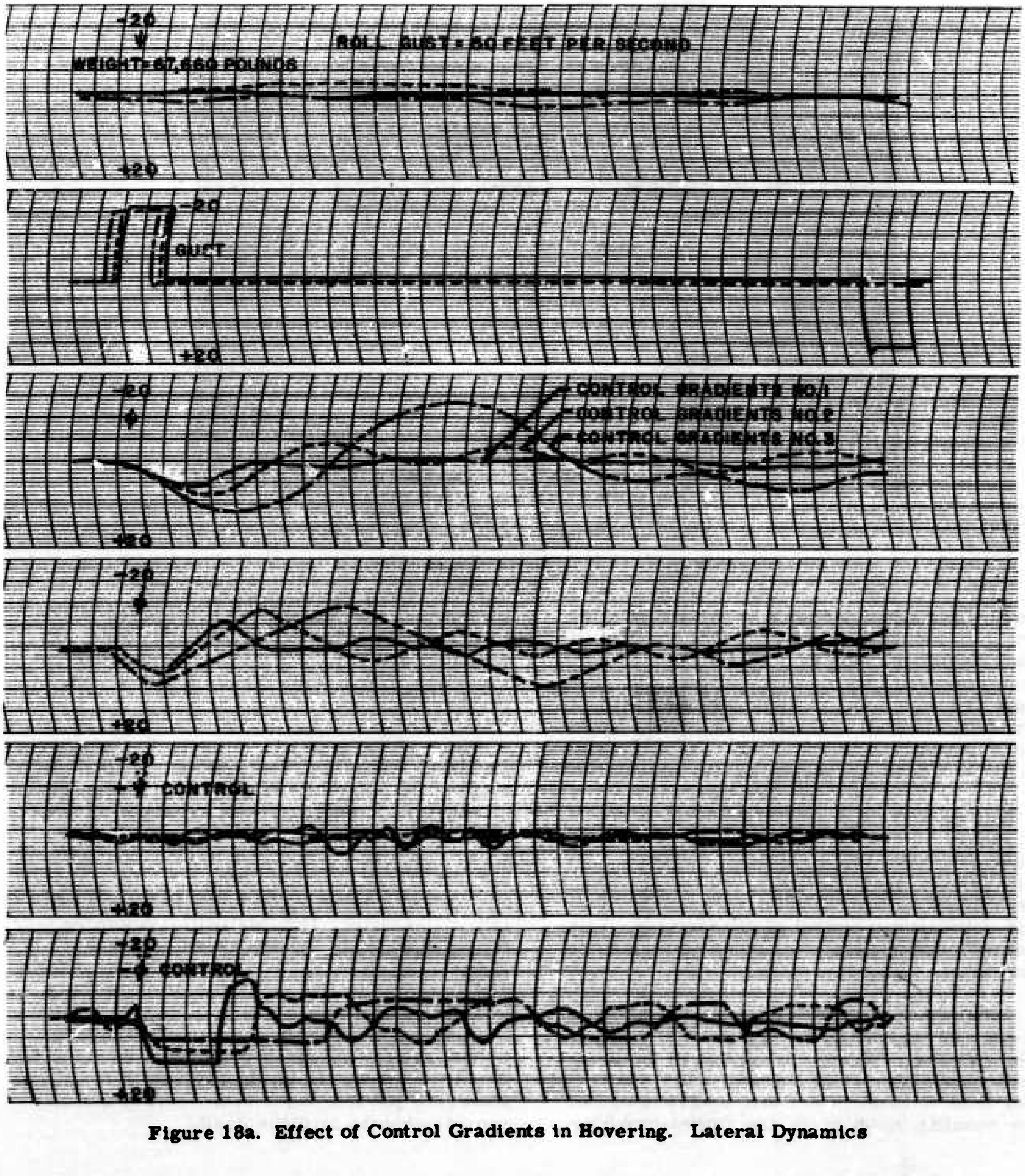


Figure 18a. Effect of Control Gradients in Hovering. Lateral Dynamics

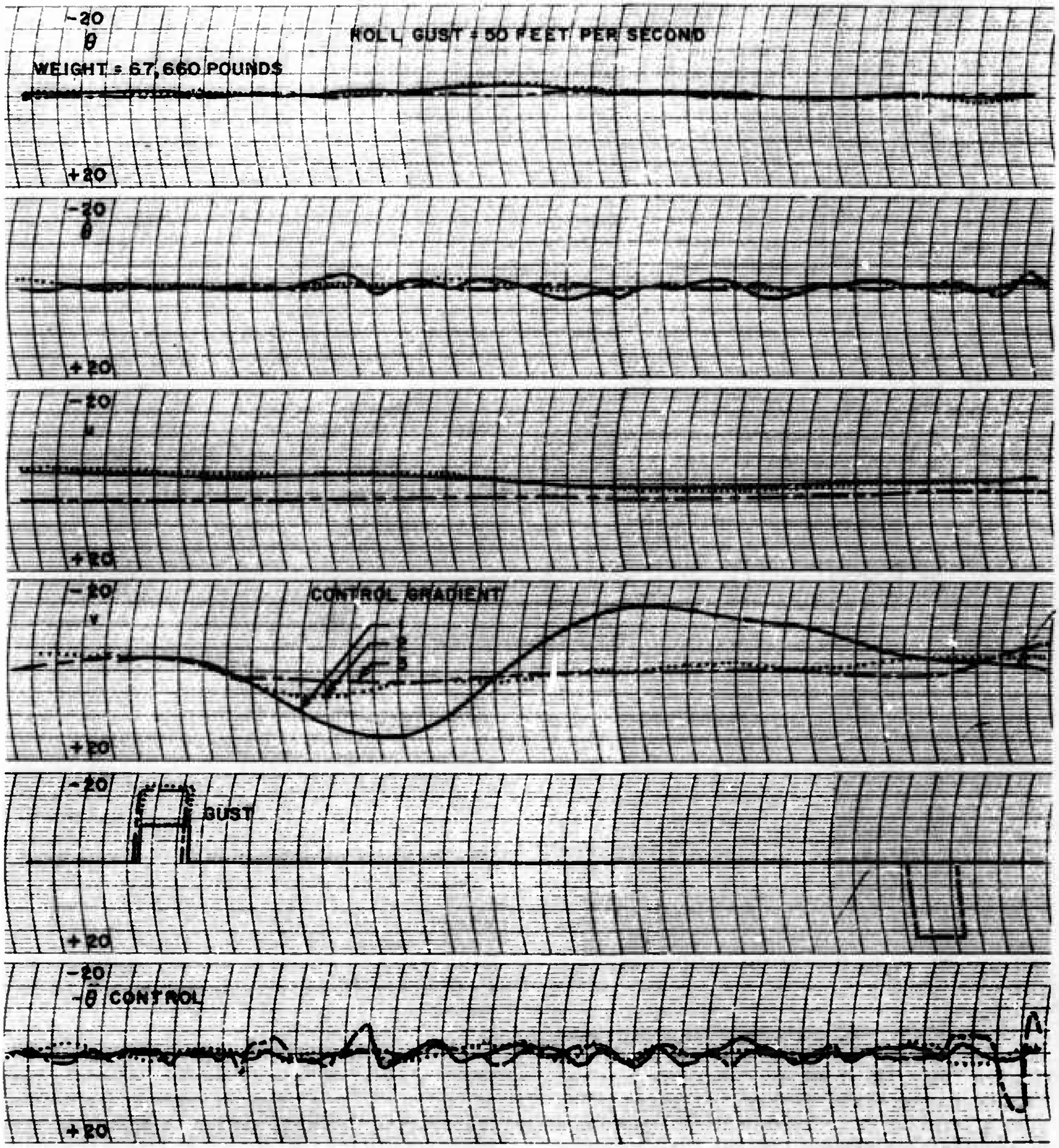


Figure 18b. Effect of Control Gradients in Hovering. Longitudinal Dynamics

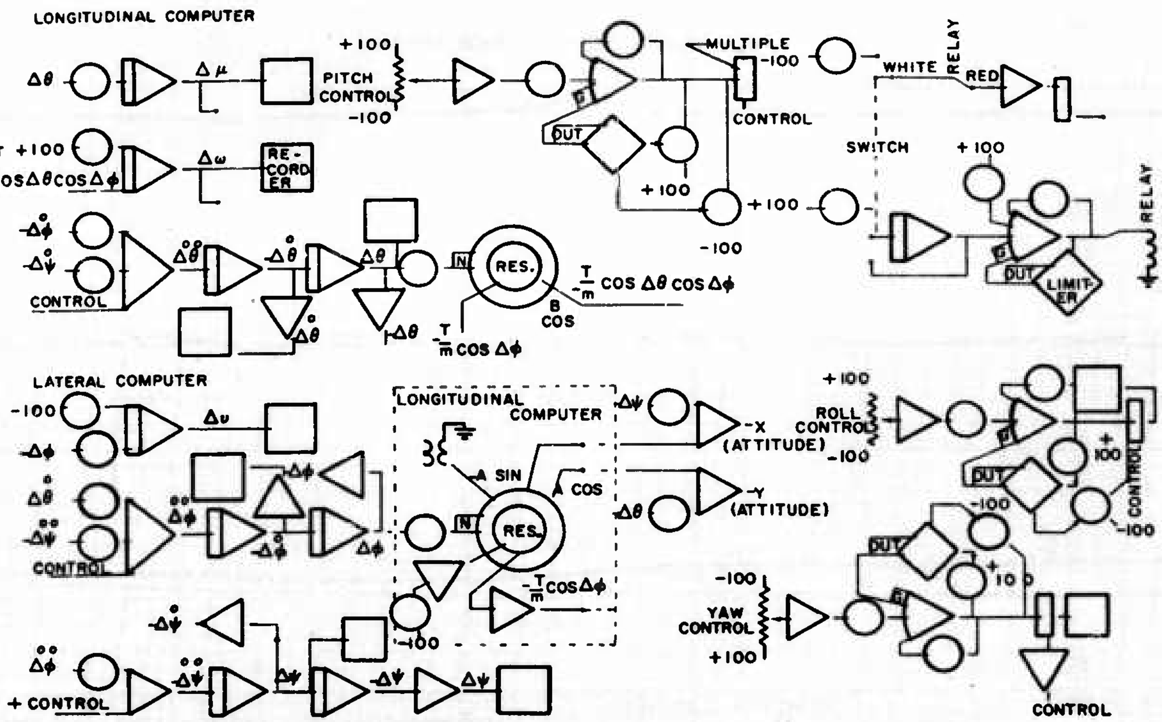


Figure 19. REAC Hovering Schematic Diagram

The three separate circuits on the right, represent the pilot's actions, applying a voltage representing the control motions. By manipulation of the potentiometers with the stick or the rudder pedals, the pilot feeds a voltage of the proportional to a control input, in the pitch, roll, or yaw airplane. The analog computers permit the application of many types of gust conditions which can disturb this configuration, and permit evaluation of the pilot's response characteristics.

From this type of analysis in different equations and the records which have been illustrated (Figure 18), the hovering control schemes have been determined as shown in Figure 20. The pilot's estimate of acceptable control gradients in angular acceleration plus the corresponding moment which is required to obtain that control gradient for this assault transport airplane are shown in Figure 21. It has been found that these moments and their corresponding gradients can be obtained by using a single J85 engine at the tail of the airplane to provide pitch control. Each outboard ducted propeller has been equipped with a split-flap arrangement which provides a thrust-spoiling effect, giveing roll control during the hovering flight. Using this split flap as a closed single flap, and operating it differentially, yaw control moments are obtained.

The detailed consideration of hovering stability control problems is an important part of any study of a vertical take-off and landing transport. Generally accepted solutions have been evalued for the particular type of transport which was studied by Bell Aircraft.

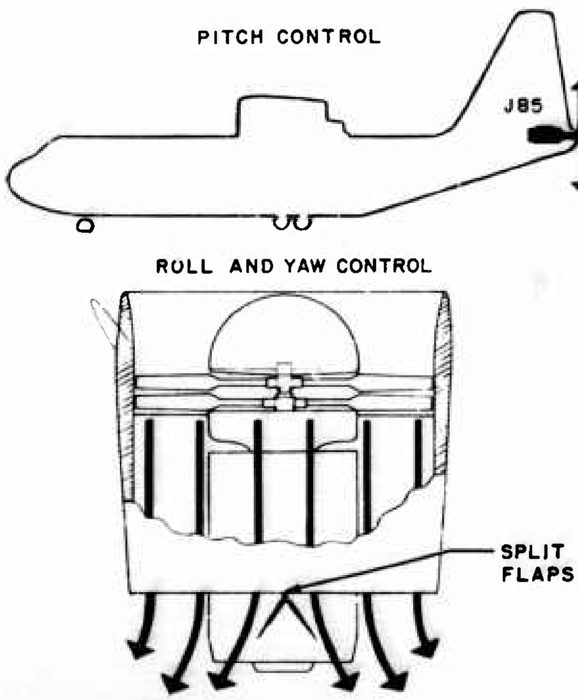


Figure 20. Hovering Control System

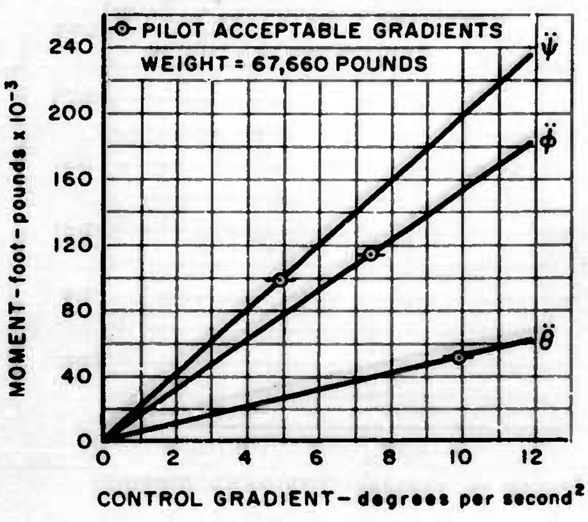


Figure 21. Reaction Control Moments Required in Hovering for Various Gradients

Conclusions

A summary of the significant results obtained during the study can be presented.

1. The application of ducted propellers yields a simplified arrangement with high static thrust, and provides a competitive assault transport solution over the speed range from zero to 527 miles per hour.

2. The airplane, to accomplish the basic mission, can be built for 67,380 pounds.

3. The performance of the configuration is excellent, with speeds up to 527 miles per hour and altitudes up to 50,000 feet. Once again the importance of speed in reducing exposure time in moving from one point to another should be emphasized.

4. All radius calculations have been based on 80% at altitude and 20% at sea level, and a mission with more than a 300 mile radius, at a speed of 455 miles per hour, can be accomplished.

5. It is possible to accomplish the 425 mile radius at an altitude below 12,000 feet.

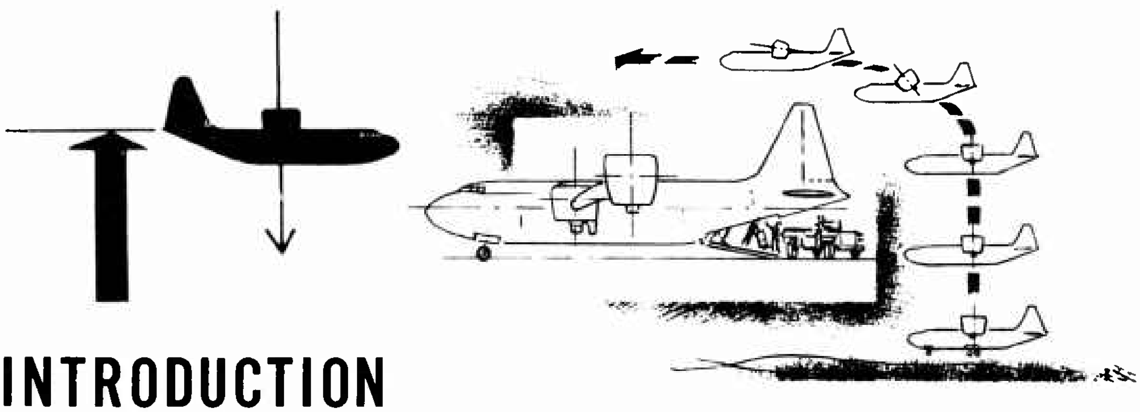
6. Using the STO/VTO combination, a basic radius of 987 miles can be obtained or loads greater than twice the basic payload of the airplane can be carried.

7. Good hovering characteristics of from 7 to 100 minutes are possible.

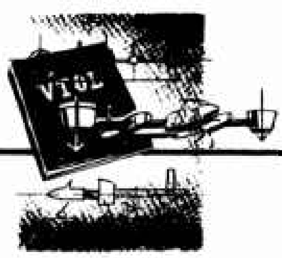
8. A full payload can be carried for a range of 2520 miles in the STO configuration.

9. The maximum ferry range of the airplane is about 3200 miles.

This study has adequately demonstrated that a VTOL ducted propeller transport is a feasible configuration. It is recommended that accelerated development of the system be undertaken at this time.



INTRODUCTION



Background

The Bell Aircraft Corporation has conducted extensive investigations of VTOL aircraft since the early post-World War II period. The early studies were directed toward high-speed "horizontal attitude" fighter-type aircraft using turbojet engines. These initial studies included USAF design study contracts on jet VTOL fighter aircraft. It was concluded from the results of this work that horizontal attitude VTOL aircraft were very feasible and practical on the basis of safety, ease of low-speed control, and operational versatility.

A tangible result of these conclusions was the design, development and construction of a VTOL Air Test Vehicle (Figure 22) by Bell Aircraft. This project was company sponsored except for the J44 engines which were obtained on bailment from the Air Force. Initial hovering tests were accomplished early in 1954 and the flight test program was continued to demonstrate feasibility of the aircraft in all spheres of hovering, transition, and conventional level flight. Practical experience in analysis, design and development of the hovering and slow-speed stability and control system was obtained in this program. Much knowledge of operational capabilities and limitations was gained in addition to the valuable design experience on a VTOL aircraft.

As result of the demonstrated feasibility of this VTOL type, and subsequent studies, the Bell Aircraft Corporation was awarded a contract to design, fabricate, and test an Advanced VTOL Airplane. This aircraft has been designated the X-14 Research Airplane (Figure 23) and is scheduled for initial flight tests in July 1956. This aircraft is powered by two Armstrong Siddeley ASV-8 Viper turbojet engines

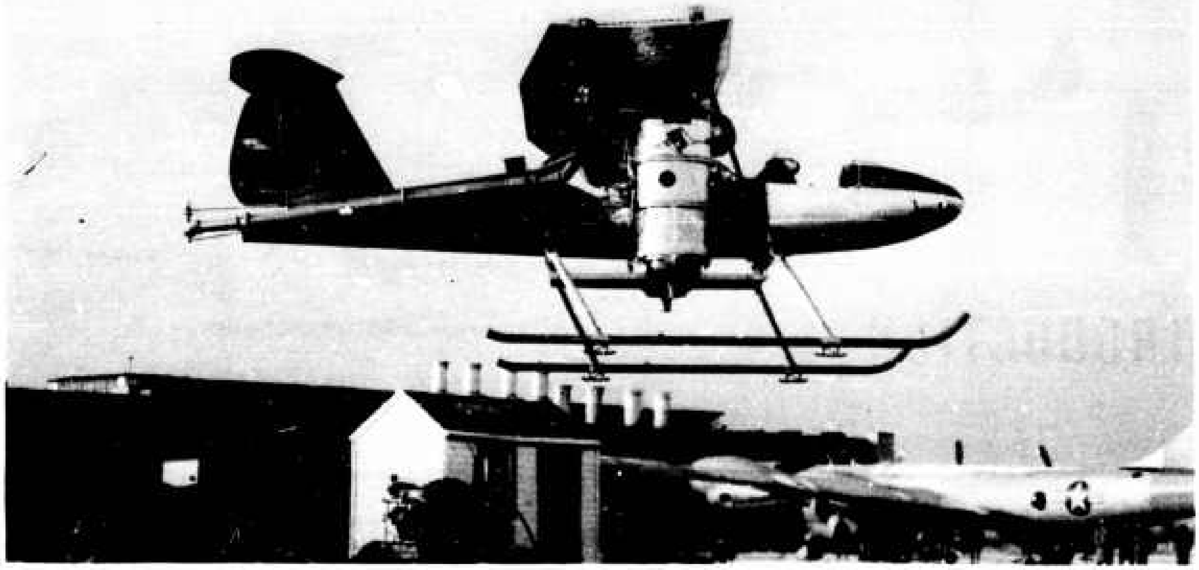


Figure 22. Bell Aircraft Air Test Vehicle

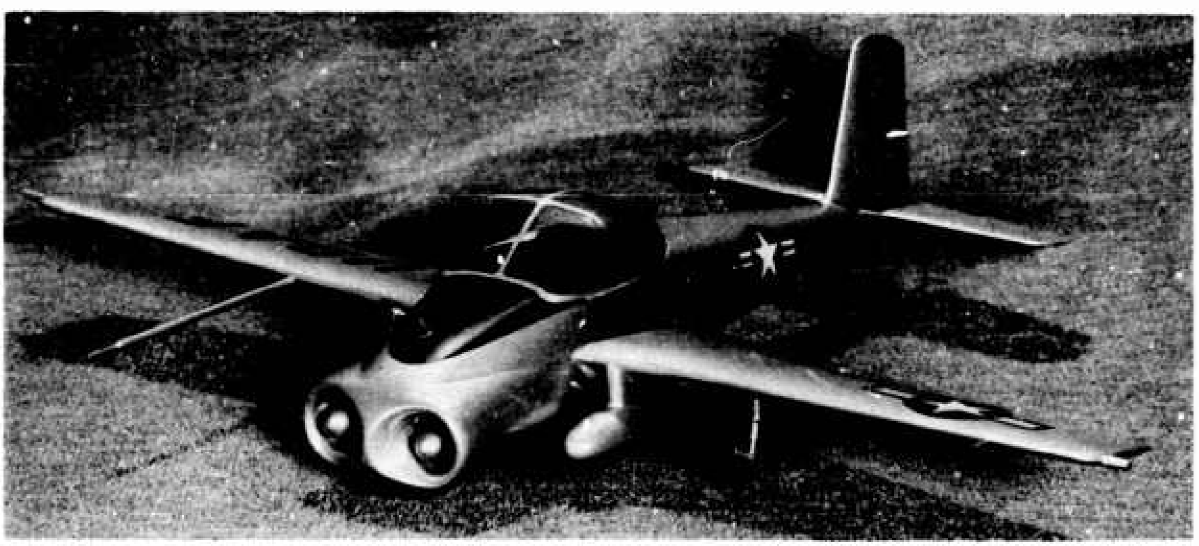
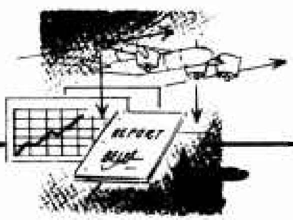


Figure 23. Bell X-14 Research Airplane

and will weigh about 2900 pounds. The additional knowledge and experience gained from this project will greatly facilitate the design of future VTOL aircraft and ensure their practicability and operational usefulness.

Recognizing the advantages and operational versatility of VTOL aircraft for applications to slower speed aircraft, Bell Aircraft investigated other methods of obtaining the vertical thrust required. Exploratory studies indicated the very favorable characteristics of the ducted propeller propulsion system. Design studies were initiated to determine the practicability of such a system. These studies revealed very suitable applications to VTOL aircraft in the subsonic speed range, such as medium haul passenger and cargo transports and lighter utility or tactical support aircraft. The obvious advantage of these aircraft was the superior economy of operation and improved hovering capabilities when compared to the turbojet-powered VTOL aircraft types.



Present Study

As result of Bell Aircraft Corporation's expression of interest in the applications of ducted propeller propulsion system, Contract Nonr-1675(00) was awarded to Bell Aircraft by the Office of Naval Research under sponsorship of the Army Transportation Corps. The purpose of the study was to investigate and evaluate the application of a ducted propeller propulsion system to the design of an Army assault transport aircraft. Work on the contract was begun on 1 May 1955, in advance of the formal contract date of 5 July 1955. The date set for completion of the contract was 15 May 1956.

The major portion of the contract has been devoted to theoretical and analytical investigations of the duct and propeller analysis, and the formulation of design methods for practical determination of ducted propeller units. The remainder of the contract was used to perform design studies of aircraft propulsion system components, and configuration studies, to determine promising aircraft designs using this propulsion system with engines which will be available in 1960. The configuration which appeared most satisfactory was selected, and a fairly comprehensive preliminary design study was accomplished to determine its physical, aerodynamic, and performance characteristics.

This report summarizes the results of the study program and briefly reviews the investigations which were accomplished during the separate phases of work. Performance comparisons of the configurations considered are presented and the results of the parametric studies are discussed. Important features relating to the aircraft systems, propulsion, structures, and weights are also included.

DESIGN CONSIDERATIONS

Requirements

The basic ground rules for the studies were determined by joint agreement of the contracting agencies and the study contractors. The aircraft design requirements used by Bell Aircraft for the study were defined as follows.

Operational Features

a. Vertical take-off and landing (VTOL) capability at ambient conditions up to 6000 feet pressure altitude at 95°F air temperature. Horizontal fuselage attitude VTOL is considered mandatory.

b. Adequate hovering stability and control of the aircraft to effect a controlled landing attitude in case of a one engine out emergency condition.

c. Aircraft availability in the 1960 to 1965 time period using engines which can be obtained prior to or during 1960.

Performance

a. 425 statute mile radius with 80% of radius at cruise altitude and 20% at sea level.

b. 300 miles per hour minimum cruise speed.

c. Zero length take-off run over a 50-foot obstacle.

d. Adequate hovering capability to satisfactorily perform limited hovering missions when required.

Operational Capability

a. 8000-pound outbound payload and 4000-pound inbound pay load.

b. Capacity for 35 airborne troops or equivalent vehicular or bulk cargo in an 8 ft by 9 ft compartment cross-section.

c. Expeditious loading and unloading to minimize ground operations, especially at the extreme radius landing site.

Basic Mission Description

The basic mission to be considered in the Assault Transport Studies was formulated from the foregoing requirements. The flight plan to perform the basic mission is described subsequently in each of its phases.

a. Vertical take-off at 6000 feet altitude and 95°F air temperatures. All landings and subsequent take-offs are vertical. Outbound payload is 8000 ounds.

b. Climb to cruise altitude and fly 80% of the radius.

c. Descend to sea level and fly the remaining 20%.

d. Land vertically at 6000 feet and 95°F. Exchange outgoing payload for 4000-pound return load. No fuel is added at the radius point.

e. Take-off vertically at 6000 feet and 95°F.

f. Return leg same as outgoing leg. First 20% at sea level, climb to cruise altitude for remaining 80%.

g. Land vertically at 6000 feet and 95°F holding a 10% total fuel reserve.

On all the range and radius calculations the installed fuel flow was increased 5% as specified in MIL-C-5011A. The basic mission profile is presented in Figure 24.

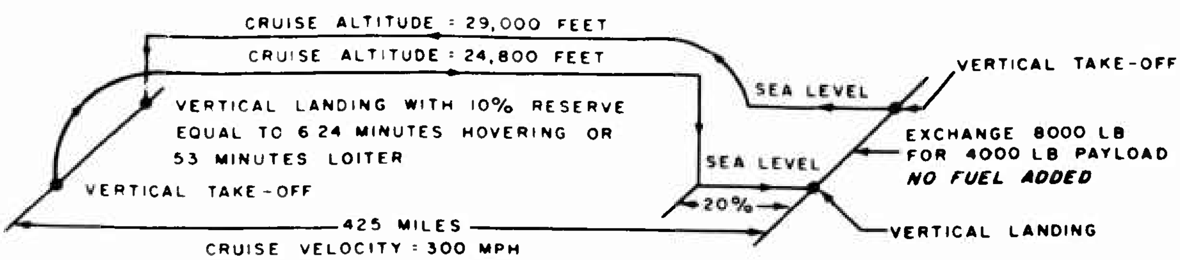
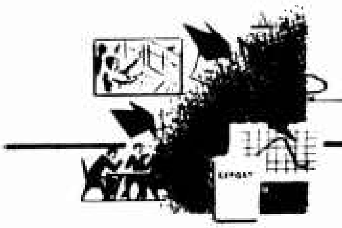


Figure 24. Basic Mission Profile



Approach

The basic method of propulsion used by Bell Aircraft in its study of VTOL assault transport aircraft was designated to be a ducted propeller system under the terms of Contract Nonr-1675(00). This decision was reached as the result of Bell Aircraft's interest in examining the possibilities of this system for application to the general area of VTOL aircraft.

Ducted Propeller Advantages

The initial exploratory studies of ducted propeller VTOL applications revealed some indication of the gains that can be obtained by using this system.

a. It is generally known that the static thrust of a shrouded propeller is at least 25% greater than that obtainable from the same diameter bare propeller with identical power input. This is illustrated in Figure 25 for theoretical conditions. In actual conditions the effect is often more pronounced.

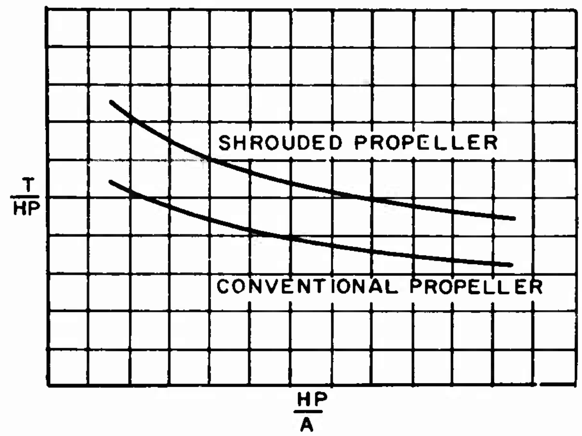


Figure 25. Static Thrust Comparison

b. The ducting around the propeller causes a reduction of the velocity effect upon the propeller as the airplane moves from static conditions to high speed flight. This reduces the pitch variation needed for efficient operation of the propeller through the whole range of airplane operating speeds.

c. Since the shrouded propeller has a smaller diameter than the equivalent bare propeller, it can be designed for high rpm, thereby relieving gearing compilation and reducing gear box weights.

d. The extreme yaw angle effects on an open propeller blade are avoided by providing a faired inlet and ducting to provide a uniform channel for air flow into the propeller. The high yaw angle effects on the propeller can cause aerodynamic forces which will complicate the stability of the configuration during transition.

e. The noise level is reduced by ducting the propeller as has been shown in NACA experiments.

f. The structural duct about the propeller is a safety feature against inadvertant personnel propeller accidents.

These considerations, coupled with a capability for reasonable cruising economy and an airplane speed potential of 300 to 500 miles per hour, made this device appear promising for subsonic VTOL applications.

Work Statement

Due to Bell Aircrafts's past experience on VTOL airplanes, a large portion of the study was concentrated on the new item of this airplane — namely, the ducted propeller. Since there was little information available on such a device, it was important to get full understanding of the capabilities and limitations early in the study. When it was felt that the ducted propeller principles and application were generally understood, they were included in the configuration design studies which were kept at a low level during the initial phases of investigation. The study of the application of ducted propeller to the assault transport requirements was organized about the following major steps:

a. Collection of the available background information on ducted propeller theory, design, and experiment, either by research or conferences.

b. Systematic investigation of the various parameters which influence the ducted propeller performance to determine the best operating range for the assault transport application.

c. Detailed analyses of propellers and shrouds to determine a combination suitable for incorporation into the aircraft configuration studies.

d. Conduct performance and stability and control analyses on the aircraft configurations chosen.

e. Determination of practical systems of power transmission from the engines to the propellers.

f. Conduct introductory studies of assault transport configurations utilizing the results of the propulsion system study.

g. Accomplish a brief preliminary design of a promising configuration of an assault transport capable of meeting the requirements designated for the study.

In actual practice the study was conducted along the paths outlined above. In addition to the accomplishment of these tasks, supporting work and liaison was conducted in connection with a ducted propeller wind tunnel program which was established by ONR at the University of Wichita.

Design Principles

Basic Concept

The study of the ducted propeller VTOL assault transport was based upon or aimed toward the Bell Aircraft concept of "horizontal attitude", rotating thrust vector VTOL aircraft which it feels is the optimum solution to the many problems of VTOL operation. With this thought in mind and in view of Bell Aircraft's extensive background in the field of jet VTOL aircraft based upon this concept, it is easy to visualize the reasoning behind the selection of the ducted propeller as a logical means of extending the horizontal attitude, rotating thrust vector concept into the subsonic aircraft field. The similarity between a rotating ducted propeller unit and a rotating turbojet thrusting device is readily apparent.

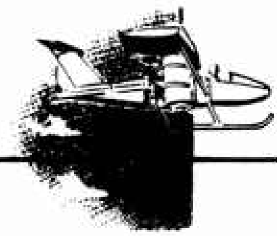
VTOL Design

In vertical take-off and landing aircraft the weight factor is of crucial importance, so that the utmost effort should be exerted to obtain a light and efficient structure taking advantage of advanced materials and techniques to achieve this end. Also, the equipment items should be selected carefully with due allowance for state of the art advances, and restricting the systems to only those items which are necessary to perform the aircraft missions.

It is recognized that the size and weight of the aircraft is, to a large extent, determined by the propulsion system which is necessary to provide the VTOL capability. Therefore, it was necessary to determine the lightest possible system consistent with reasonable development, fabrication, and maintenance of the components. In connection with this, the use of turbine propeller power plants is considered necessary to the success of any ducted propeller VTOL transport.

The vertical take-off capability places greater emphasis upon the importance of center of gravity control and the location of the resultant vertical thrust vector, so that adequate stability and control characteristics will exist when the aircraft is in the hovering phases of the flight plan. In addition, satisfactory aircraft attitude control must be maintained in the event of partial power failure during hovering so that a controlled attitude ground impact can be negotiated.

The additional versatility gained by the short take-off operation capability of the horizontal attitude VTOL aircraft, is an important factor to be considered in the concept and design of the aircraft. In addition, structural criteria, loads and analysis methods must be modified or initiated for use with the combination VTOL - STO types which are investigated.



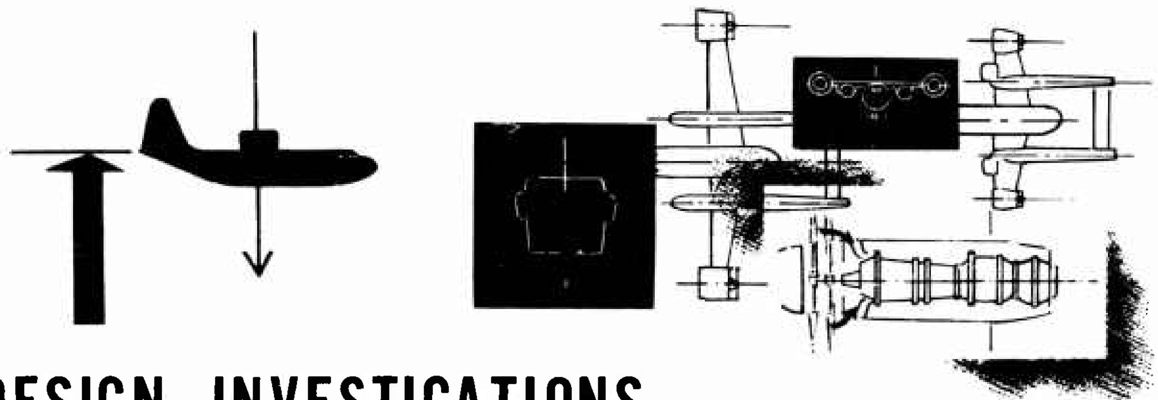
Experimental Program

Wind Tunnel Program

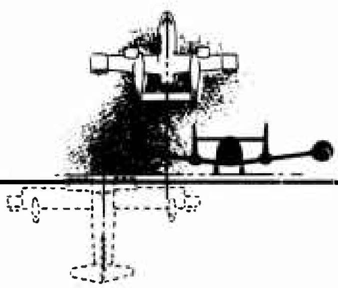
During the pre-contract discussions, Bell Aircraft technical personnel recommended the establishment of an experimental wind tunnel investigation of ducted propeller characteristics. It was known at that time that very little work on ducted propellers had been conducted in the United States and that there had been a very small interest in the field during recent years. Therefore, it was suggested that a program be organized to obtain basic data on the design of ducted propeller units suitable for application to transport type aircraft.

Research Aircraft

The research aircraft projects undertaken by Bell Aircraft has proven the value of such programs. Among these have been the successful jet VTOL aircraft which were so valuable in proving the feasibility of the Bell concepts in the field. It is believed a small ducted propeller aircraft will be of great value in proving the feasibility of the ducted propeller propulsion system and in obtaining practical operating data and experience on the aircraft type. Without a doubt, the operational characteristics will suggest problems, advantages, and useful applications of ducted propeller aircraft. A research on test bed airplane program would be a logical step in the development of any aircraft which will use this propulsion system.



DESIGN INVESTIGATIONS



General Configuration Studies

General

The major design factors considered in the configuration studies, were contained in the requirements set out as the ground rules at the inception of the study. The aircraft size in this case was determined by the basic payload required and the cargo compartment cross-section dimensions desired. These were 35 airborne troops weighing 8000 pounds or equivalent cargo to fit the 8 feet by 9 feet compartment dimension.

The vertical take-off requirement exerted a great influence upon the design concepts. First, the thrusting units must be arranged so that the resultant should pass through the airplane center of gravity. Secondly, this requirement will allow the use of higher wing loadings which are better suited to the airplane cruise conditions. Also, the need for flaps to increase lift coefficient at landing is eliminated, thus leading to more efficient and lighter wing structure.

The obvious safety requirement of airplane attitude control with partial power failure during hovering, influenced the propulsion system design so that uncontrollable moments would not be introduced under these conditions.

In addition, the operation of the aircraft as a short take-off vehicle was considered from the outset, so design requirements of the landing gear and aircraft structure were included.

Twin-Duct Configuration

In the early stages of the study a preliminary two-duct configuration was established to be used as a basic design for aerodynamic analyses of a typical ducted propeller transport aircraft. The basic propulsion units for this design were established in the initial aerodynamic ducted propeller studies. Preliminary were to be conducted for this configuration. The design parameters chosen for the design were 48,000 pounds gross weight, 8000 pounds payload, and 15,000 pounds fuel. A general arrangement of the configuration is presented in Figure 26. It must be emphasized that this configuration was established as a working tool from which would evolve other more refined and realistic assault transport designs.

This configuration was used as a point of departure for the first aerodynamic studies. It is only reasonable that this configuration was also used as the subject of the first intensive design studies and preliminary structural analyses. It was realized that the configuration was based upon very rough assumptions, especially with regard to the propulsion and control systems, and there was a good likelihood of an incompatible design. Nevertheless, the information obtained from an integrated study of a single configuration can be very valuable as basic data in the determination of later more realistic designs.

The description of the propulsion system design study for this aircraft is contained in another section of this report. Two Wright T49 turboprop engines were selected for the power plants and the power was shafted at high speed to the outboard gear boxes. Freewheeling clutches at each engine gear box and an interconnecting shaft insures division of power to the two propellers.

The initial detailed duct and propeller studies were carried out for this configuration (Reference 2), a satisfactory aerodynamic design was accomplished, and a system using the physical characteristics so determined was designed in a preliminary fashion. A tenth scale model of this design was also specified for test in the University of Wichita subsonic wind tunnel. A weight analysis of the full size system designed is summarized in Table III. These data are considered to be conservative

TABLE III. PROPULSION SYSTEM WEIGHTS FOR TWIN-DUCT T49-POWERED CONFIGURATION

		Weight(pounds)
Duct Structure (Nacelle Section)		
Center Shell Including Leading Edge Flaps and Actuators		850
Center Body Structure		203
Exit Stators		99
Inlet Guide Vanes		136
Support Struts		88
Actuating Mechanisms		60
	Total per side	1436
Rotating Components		
Engine Gear Box		486
Shafts		440
Propeller Gear Box		1345
Oil System		75
	Total per side	2346

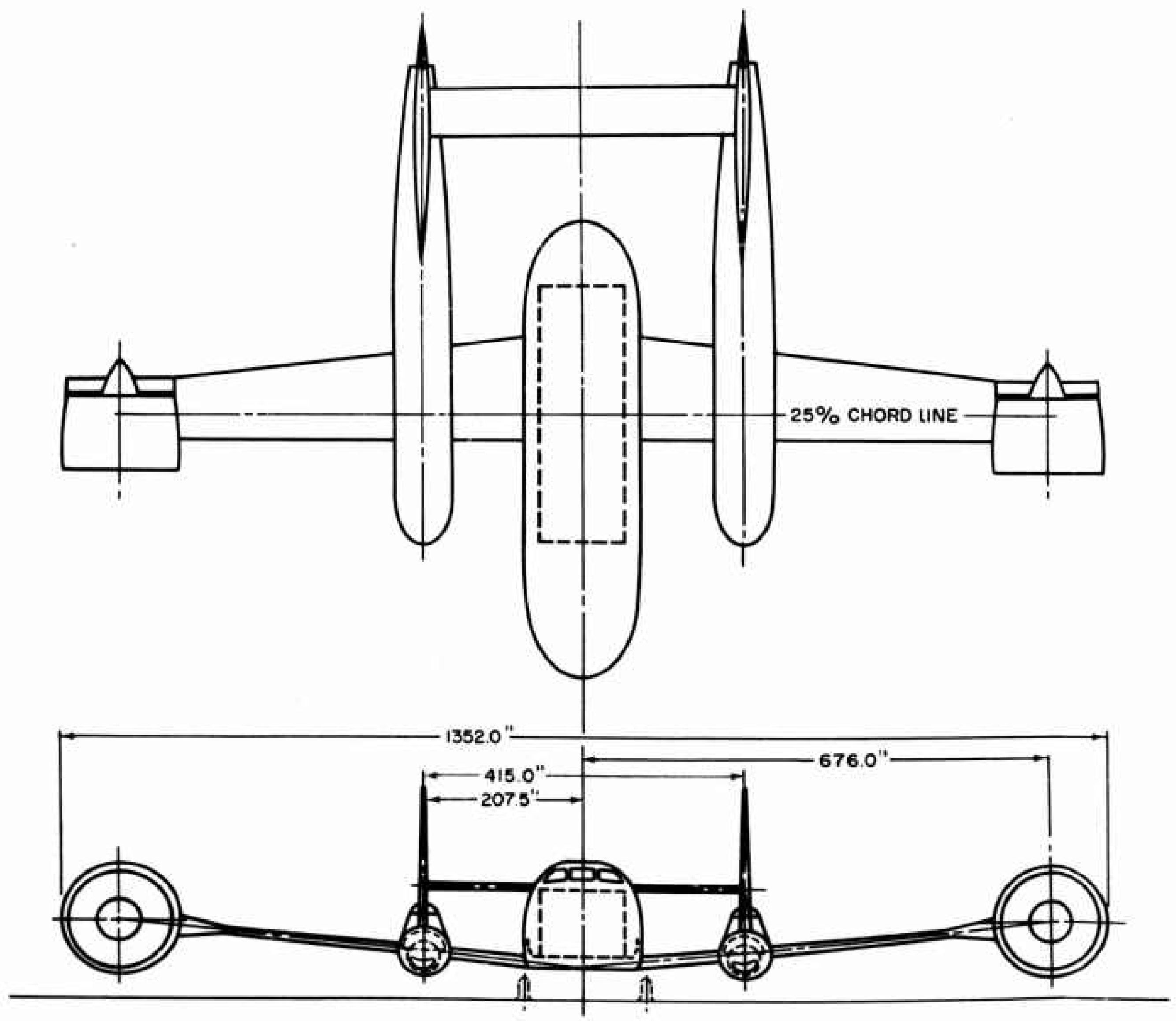
after comparison with available data on similar items (Reference 3).

The design study was continued with this propulsion system data and other information gathered from sources. The final weight results are summarized in Table IV. The design is not consistent in that the propulsion system was designed for a 50,000-pound take-off weight so the surfaces and landing gear were also sized for this gross weight. In other words, a compatible design would be much heavier than the 59,000-pound aircraft shown.

TABLE IV. WEIGHT ESTIMATE OF TWIN-DUCT PRELIMINARY STUDY CONFIGURATION

D181-960-001

		Weight (Pounds)
Wing		3500
Tail		
Horizontal		350
Vertical		600
Body		
Fuselage		3820
Booms		880
Landing Gear		1500
Surface Controls		500
Engine Sections		2000
Ducts for Fan and Wing Tips		2872
Propulsion Group		
Engine Installations - Two T49's		6980
Gear Boxes and Drives		4692
Engine Accessories		600
Lubricating System		210
Fuel System		462
Engine Controls		30
Starting System		125
Propeller Installation		1930
Auxiliary Power Plant		100
Instruments		175
Hydraulic		300
Electrical		800
Electronics		500
Furnishings		644
Air Conditioning and Anti-Ice		500
	Total Weight Empty	34,070
Crew (3)		690
Pay load		8000
Fuel		15000
Oil		240
Water-Water Injection System		1000
	Total Useful Load	24,930
	Gross Weight	59,000



TWIN DUCT VTOL TRANSPORT

WING	
SPAN	76.0 FEET
AREA	885.0 SQUARE FEET
ASPECT RATIO	6.52
FUSELAGE AND BOOMS	
FUSELAGE LENGTH	47.5 FEET
FUSELAGE DIAMETER	13.3 FEET
NACELLE DIAMETER	3.8 FEET
BOOM LENGTH	38.8 FEET
TAIL	
VERTICAL AREA (EACH)	100.0 SQUARE FEET
HORIZONTAL AREA	178.0 SQUARE FEET
PROPULSION SYSTEM	
ENGINES (2)	WRIGHT T49 TURBOPROP
DUCTS (2)	
PROPELLER DIAMETER	11.6 FEET
DUCT O.D.	12.5 FEET
DIAMETER WITH EXTENDED INLET FLAPS	15.5 FEET
HUB DIAMETER / PROPELLER DIAMETER	0.4

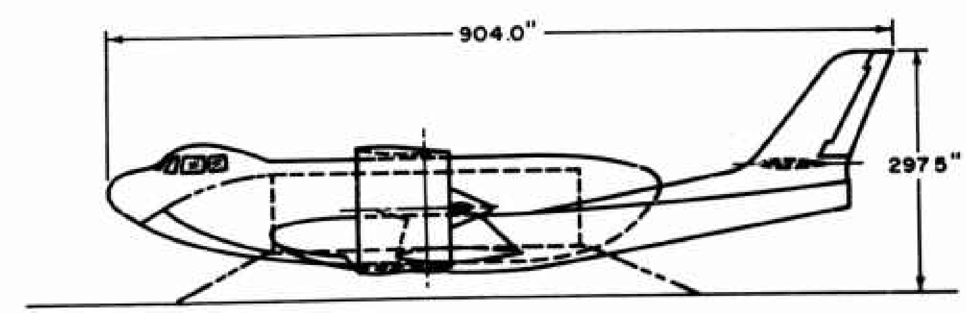


Figure 26. Twin-Duct Assault Transport with T49 Engines

Four-Duct Configurations

The twin-duct configuration had been designed around the minimum size cargo compartment to carry the specified 8000 pound payload. When the aerodynamic performance studies (Reference 4) indicated increased pay load capabilpties of STO operation, it was decided that a larger cargo compartment should be used to accommodate the increased STO pay loads. Therefore, the aircraft size was somewhat increased to obtain a more versatile over-all design.

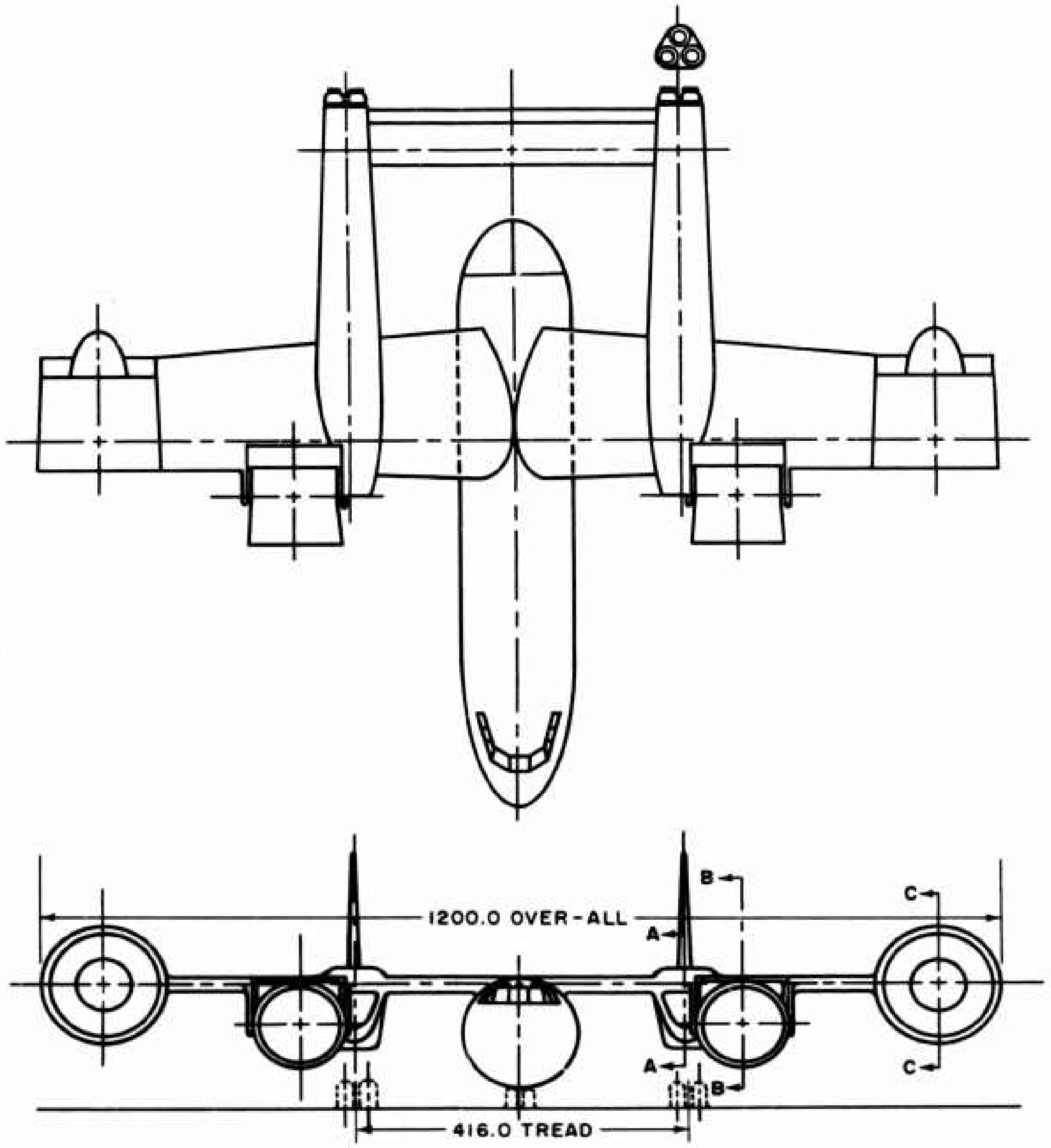
The estimated gross weight of the larger configuration required a duct diameter for a two-duct configuration which appeared too great for adequate ground clearance angle. It was felt that a four-duct configuration would relieve this situation, and investigation of a four-duct configuration was judged desirable to obtain more information for the general power-plant system design study. The first four-duct configuration studies were based upon propulsion systems in which a pair of coupled turboprop engines drove the large outboard ducted propellers and a single engine of the same type powered the inboard propeller. A typical design based upon the use of the Allison 550-B1 engines is presented in Figure 27.

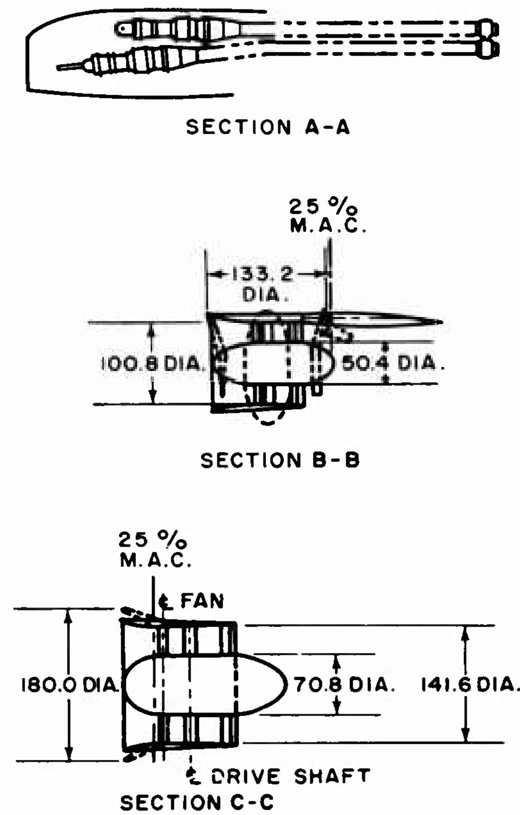
The use of six engines occasioned a departure from the basic concepts of two-engine operation. Mechanical interconnecting of the ducts was eliminated, since the multiple engine installation allows maintenance of thrust balance by engine power manipulation in cases of single engine failure. That is, the loss of power from one of the Allison 550 units could be compensated by immediate shutdown of the corresponding unit on the opposite wing, or by partial reduction of power of the engines on the opposite wing to maintain thrust symmetry of the propeller units. The shaft lengths were kept to the minimum, and the lowest possible number of gear boxes were specified for the configuration. These criteria resulted in a design which groups the engines in nacelles on each wing, and the tail pipe extensions give the aircraft a twin boom type of configuration. An estimated weight summary is presented in Table V.

Four-Duct Tilting Engine Configurations

In the study of four-duct configurations, the concept of propeller units incorporating the turboprop engine in the duct centerbodies was investigated. One disadvantage of this arrangement is the necessity for the engines to operate at angles of tilt up to 110°. Another requirement would be the need for an auxiliary reaction control system for hovering and slow speed operation. The advantages of the system are the elimination of shafting, and the contribution of the residual jet exhaust to the lifting thrust of the ducted propeller units. Six Westinghouse RB109 turboprop engines are used in this application located in the duct centerbodies. A general view of this configuration appears in Figure 28, the general arrangement drawing showing the over-all configuration. This was based on a rough weight and balance analysis from which placement of major aircraft components, determination of surface areas, and location of the propulsion units were accomplished. In the course of this work, it was found that the installation of a single General Electric J85 engine in the aft fuselage would be adequate to furnish pitch reaction control during hovering and slow speed flight. The propulsion units were disposed longitudinally so that the resultant thrust vector would act through the airplane center of gravity. A typical military transport fuselage mounted landing gear arrangement has been considered, and the normal 48-inch cargo compartment floor height is a design feature.

During the general arrangement study, the combination of the duct, propellers, and engines was considered. With the engines located in the duct centerbody, the problems of power transmission have been appreciably reduced. For the inboard ducts, the propeller is driven directly from the engine output shaft. In this case, it is assumed that the existing engine gear box has been redesigned to delivery the required propeller rpm. The pivot point of rotation for the unit is so located that adequate ground clearance is provided when the duct is in the vertical take-off position. The larger outboard ducts are designed to contain contrarotating propeller units powered by the two turbine engines in the duct centerbody. The





DUCTED PROPELLER VTOL TRANSPORT
ALLISON 550-BI ENGINES (6)

VERTICAL TAIL (2)

AREA-EACH	123 sq ft
ASPECT RATIO	1.7
M. A. C.	8.5 ft
SECTION	NACA 0.0009

HORIZONTAL TAIL

AREA	214 sq ft
ASPECT RATIO	5.6
M. A. C.	6.2 ft
SECTION	NACA 0.0009

POWER PLANT

ENGINES (6)	ALLISON-550-BI
HORSEPOWER-(TOTAL T.O.)	31,008

WEIGHTS

GROSS	75,000 lb
EMPTY	50,000 lb
CARGO	8,000 lb
CREW (3)	600 lb
FUEL	15,000 lb
OIL	400 lb

WING

AREA (NEGLECTING DUCTS)	1,082 sq ft
A.R. (NEGLECTING DUCTS)	5.08
M.A.C.	14.59 ft
SECTION	NACA 64A-412

DUCTS

OUTBOARD (2)	MAX. O.D. (INLET NOSE)	15.0 ft
	PROP. DIA.	11.8 ft
	INNER BODY DIA.	5.9 ft
INBOARD (2)	MAX. O.D. (INLET NOSE)	11.1 ft
	PROP. DIA.	8.4 ft
	INNER BODY DIA.	4.2 ft

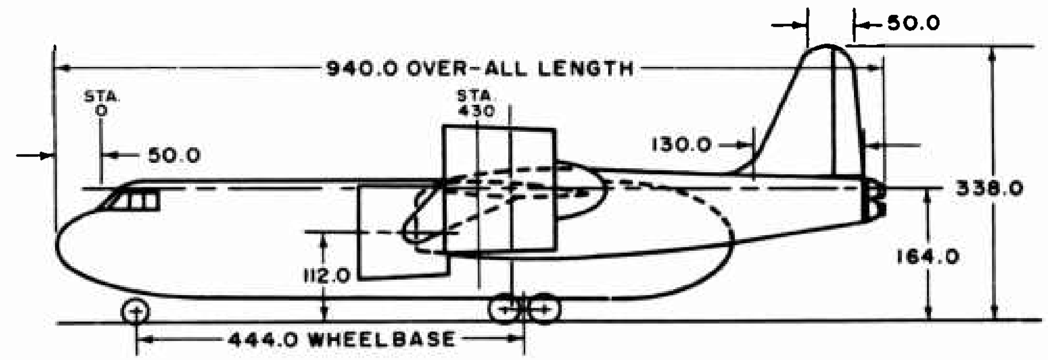


Figure 27. Ducted Propeller Twin-Boom Medium Cargo Transport

TABLE V. TWIN-BOOM ALLISON 550 ASSAULT TRANSPORT D181-960-011

Item		Weight (pounds)
Wing		5,400
Tail		
Horizontal		492
Vertical		566
Body		
Fuselage		5,175
Booms and Nacelles		3,720
Landing Gear		2,220
Surface Controls		
Flight Controls		500
Reaction Controls		1,000
Engine Section (Duct Around Propellers)		
Inboard		2,780
Outboard		3,920
Propulsion		
Engines		9,450
Gear Boxes and Engines (2)		2,850
Gear Box and Inboard Propellers (2)		2,000
Gear Box and Outboard Propellers (2)		2,000
Engine Mounts		370
Duct Supports - Inboard		300
– Outboard		500
Rotating Mechanism – Inboard		60
– Outboard		100
Lubricating System		195
Fuel System		460
Water Injection System		200
Engine Controls		50
Starting System		150
Propeller Installation – Inboard		1,094
– Outboard		1,522
Auxiliary Power Plant		80
Instruments		160
Hydraulics (Brakes and Nose Wheel Steer.)		50
Electrical		800
Electronics		500
Furnishings		465
Air Conditioning and Anti-Icing		500
Auxiliary Gear (Jacking, Towing)		25
	Total Weight Empty	49,654
Useful Load		
Crew (3)		645
Oil – Engines		160
– Gear Boxes		240
Fuel		15,000
Water		1,297
Pay load		8,000
	Total Useful Load	25,342
	Gross Weight	74,996

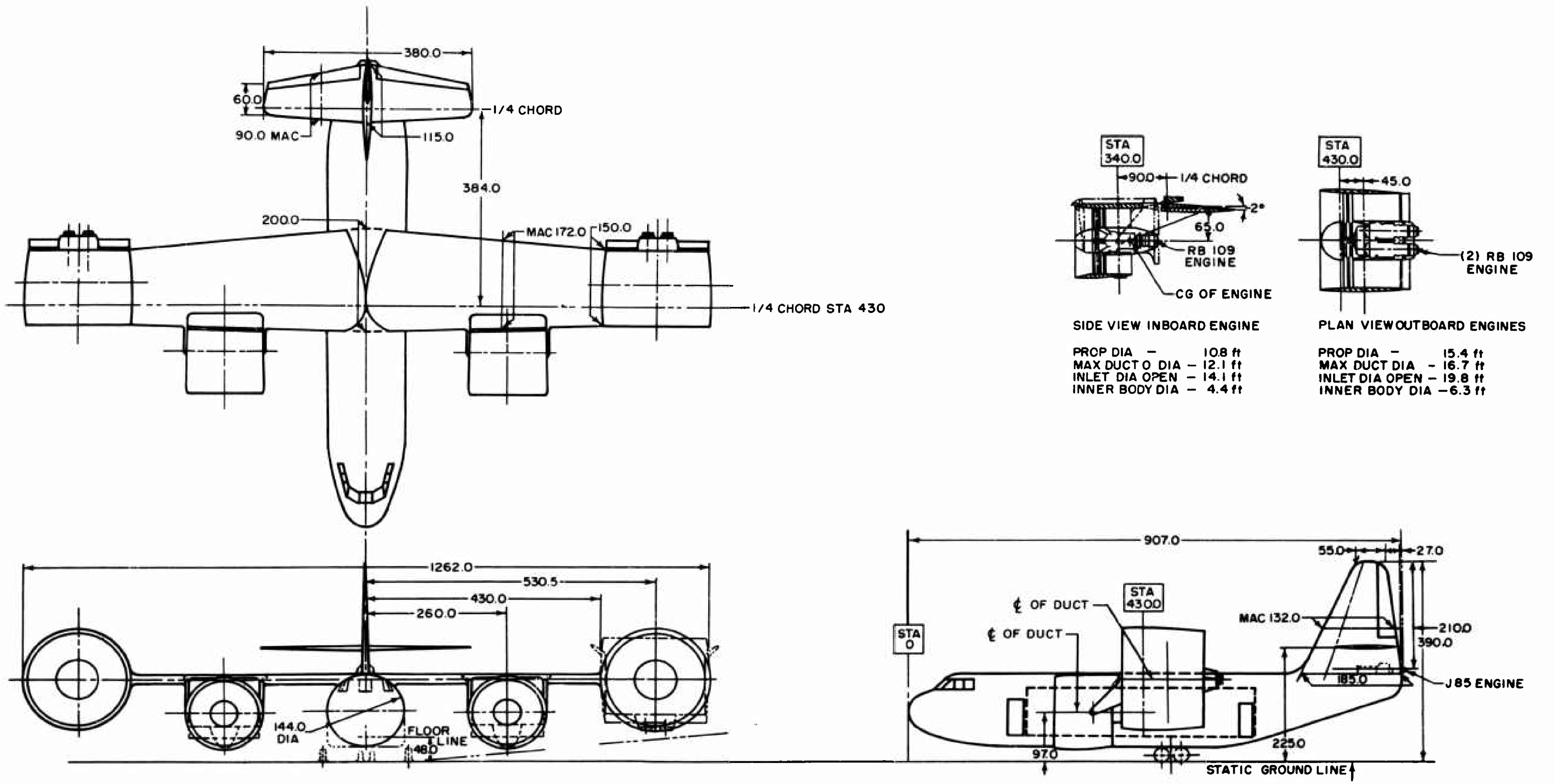


Figure 28. Ducted Propeller Medium Cargo Transport

TABLE VI. FOUR-DUCT RB109 TILTING ENGINE CONFIGURATION D181-960-007

		Weight pounds
Wing		5,200
Tail		
Horizontal		685
Vertical		486
Body		7,423
Landing Gear		
Nose		400
Main		1,900
Surface Controls		
Flight		500
Reaction Controls (Pitch)		400
Engine Section (Duct Around Propeller)		
Inboard		3,200
Outboard		5,050
Propulsion		
Inboard Engines (2) RB109		3,700
Outboard Engines (4) RB109		7,400
Inboard Engine Gear Box (Incl. in Eng.)		----
Outboard Eng. Gear Boxes (2)		1,544
Engine Mounts – Inboard		150
– Outboard		300
Duct Supports – Inboard		360
– Outboard		600
Rotating Mech. – Inboard		75
– Outboard		125
Lubrication System		195
Fuel System		460
Water Injection System		200
Engine Controls		50
Starting System		150
Propeller Installation – Inboard		1,200
– Outboard		2,200
Auxiliary Power Plant		80
Instruments		160
Hydraulics (Brakes and Nose Steering)		50
Electrical		800
Electronics		500
Furnishings (No Paratroop Seats)		465
Air Conditioning and Anti-Ice		500
Auxiliary Gear (Jacking, Towing)		25
	Total Weight Empty	46,533
Useful Load		
Crew (3)		645
Oil – Engines		188
Gear Boxes		125
Fuel		13,212
Water		1,297
Pay load		8,000
	Total Useful Load	23,467
Gross Weight (VTOL Position)		
Total Weight Empty		46,533
Useful Load		23,467
	Gross Weight	70,000
Gross Weight Less Fuel		56,788
Gross Weight Less Fuel and Pay load		48,788

output shafts from the engines, drive through a common gear box designed to drive the coaxial contrarotating propellers.

During the work on the four-duct configurations, more detailed information was developed for the fixed equipment items and aircraft systems. This data was incorporated in the weight estimate of this configuration and appears in Table VI. The application of the Allison 550-B1 engines to this configuration was undertaken in the natural course of events and selected as the best representative example.

Four-Duct Allison 550-B1 Configuration (D181-960-009)

As mentioned previously, a configuration was designed using Allison 550-B1 turboprop engines. From the outset it was thought that the use of this engine would result in a lighter, higher performance aircraft. The more powerful engines enabled the designer to reduce the duct size for the same gross weight. It is expected that the smaller duct diameter and higher propeller speed will result in lower propulsion system weight. The improved specific fuel consumption would tend to maintain aircraft endurance to roughly the same value for an unchanged fuel capacity, while the maximum performance could be expected to increase.

A design study was undertaken to determine a practical configuration. It was decided that much of the configuration using the RB-109 engines could be used for this study. As a result, the initial step was to replace the ducted fan units with new ones containing the Allison 550 engines. The wing, fuselage, empennage, landing gear, etc. remained unchanged. The result of this study is shown in Figure 29. The possible reduction in duct size is obvious at a glance when compared to the RB-109 configuration.

A comparison of the weight of this configuration with that of the RB-109 design is of interest. The gross weight of the aircraft has decreased to about 67,380 pounds, (Table VII), a difference of 2620 pounds. This could be utilized for increased payload or a larger fuel load for longer range performance.

The performance comparison of these two configurations revealed the great improvement possible by installation of the Allison 550-B1 engines. This performance gain was the major factor in the choice of the configuration as the representative result of the over-all study.

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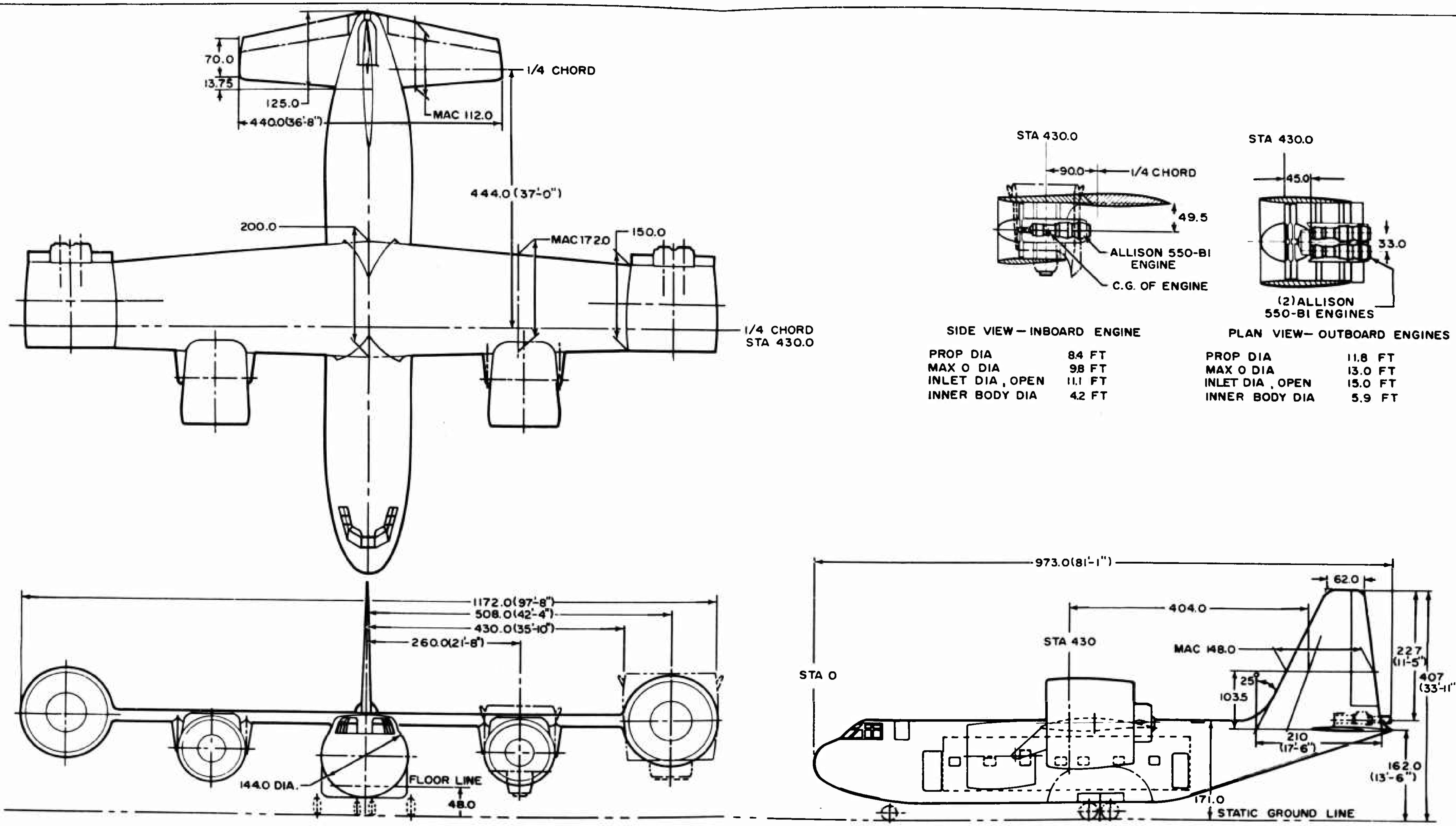

Figure 29. Assault Transport Configuration

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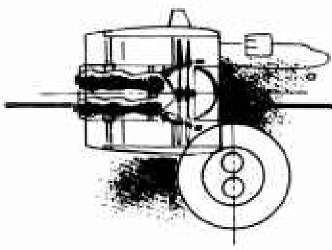


Propulsion System Studies

General

At the onset of the study, it was evident that the major unknown quantity in the design of a practical assault transport aircraft was the ducted propeller propulsion system. A limited amount of design data was available from the original brief studies which preceded the contract period.

The turbine-propeller powerplant is considered necessary to the success of any ducted propeller VTOL transport, and any improvement in the engine characteristics is immediately reflected in better aircraft performance or increased pay load capability, or allows the design of a lower gross weight aircraft. The requirement of take-off operation at 6000 feet altitude on a 95°F day is also a factor which must be considered in the selection of the power plant.

The approach to the design of the propulsion system was directed by the progress of the aerodynamic analytical studies of ducted propeller units. While the basic investigations were in progress, design studies of an exploratory nature were conducted to obtain trend data on the variation of system characteristics. Then, as soon as initial design data became available, a preliminary study configuration was selected as the object of more detailed investigations. The design knowledge, gained from the propulsion system work on this preliminary configuration, was then applied on the design studies of other configurations which eventually resulted in the final aircraft system arrangement.

Propeller Drive System Study

The original studies of the ducted propeller aircraft were based upon a ducted propeller unit at each wing tip. However, the results of a brief parametric study of possible drive systems indicated a decrease in total propulsion system weight when a number of smaller ducted propeller units of equivalent total area are substituted for the twin ducts. An aircraft and drive system arrangement was briefly investigated in which the duct units were grouped in four pairs, a pair at each wing tip and a pair under each inboard wing panel. This study resulted in a complex system of gear boxes and interconnecting shafting, and the conclusion that the problems encountered in designing, developing, and maintaining the system would more than offset the relatively small weight saving derived.

Twin-Duct System Design

A preliminary two-duct configuration was established to be used as a basic design for aerodynamic analyses of a typical ducted propeller transport. Since it was desirable to obtain design data on a twin duct propulsion system, the same configuration was also adopted as the subject for an intensive propulsion system design study. The general arrangement of this study configuration was presented in Figure 26.

The duct and propeller design requirements established by the Aerodynamic analyses were integrated into the drive system study. The propeller diameter was established by practical ground clearance requirements and the propeller blade tip speed, set by aerodynamic considerations, determined the propeller rotational speed. The powerplant selected for the configuration was the Wright T49 turboprop engine with a basic shaft speed of 8000 rpm. The weight of the interconnect shafting can be held to a minimum by transmitting the power at a high rpm. Therefore, the standard T49 reduction gearing was replaced by a single one-to-one ratio spiral bevel pair which would transmit the power to the outboard locations through the lightest practical shafting system. A section view of the engine gear box appears at the right side of Figure 30. This unit was designed for assembly on the basic engine with the reduction gear box removed. A sprag-type overrunning or free-wheeling clutch is installed to allow single engine operation of both propeller units through the interconnect shaft (bottom of gear box). Spherical couplings allow angular motion of the shafts with respect to the gear box thus preventing extraneous distorting loads on the gear casing.

The complete propeller drive system is presented in the sectional view. The power is transmitted by the torque tube into the duct gear box which reduces the speed through a right angle bevel pair and a single-stage of planetary gearing to the desired propeller speed. The system was designed fairly complete in detail to obtain reasonable weight estimates for typical drive systems (see Section IV.A.2.).

Four-Duct System Design

In order to round out the picture of powerplant system design, several four-duct configurations were studied. In these arrangements a pair of coupled engines were used to drive an outboard propeller and a single engine drove a mid-wing mounted duct propeller (Figure 9). The six engines were not interconnected mechanically so a catastrophic situation will not accompany a single engine failure in this case. In these studies, as before, it was found that appreciable weight savings were realized by transmission of power at the basic engine speed and the number of gear boxes in the system could be held to a minimum.

The simplest and most direct approach to the drive system problem, was found to be the conventional engine gear-box-mounted propeller. Although there are certain complications due to location of the engines in the duct centerbodies, the drive system weight and complexity are reduced to the practical minimum. It was found that a coupled engine could be installed in the outboard duct centerbody and a single engine would drive the inboard propeller. These designs are referred to, as the four-duct tilting engine configurations, examples of which were seen in Figures 28 and 29.

Duct Design and Inlet Flaps

The basic duct section was chosen to be a constant area channel from the propeller station back to the duct exit. However, results of aerodynamic studies have indicated the necessity of variable duct inlet area, to maintain propeller efficiency through the required range of forward flight speeds. The design study of this feature for the full-size duct was based upon the concept of modifying the leading edge of the duct profile.

The ideal inlet shape for the duct at static operating conditions is the bell-mouth or nozzle contour. For flight operation, the most efficient inlet will have a thin lip forming a nonconverging duct or even a diffuser section. To obtain efficient propeller operation at static conditions and during forward flight, it may be necessary to provide a variable position leading edge to form suitable inlet shapes at the various operating conditions.

Initial studies have shown the impracticability of flaps, which could be folded back against the exterior surface of the duct, chiefly because of the incompatible curvature of the flap segments and the duct surface.

The feasibility of retractable flapped leading edge extensions was briefly studied with the result that a tracked flap extension system similar to a Fowler flap device was designed.

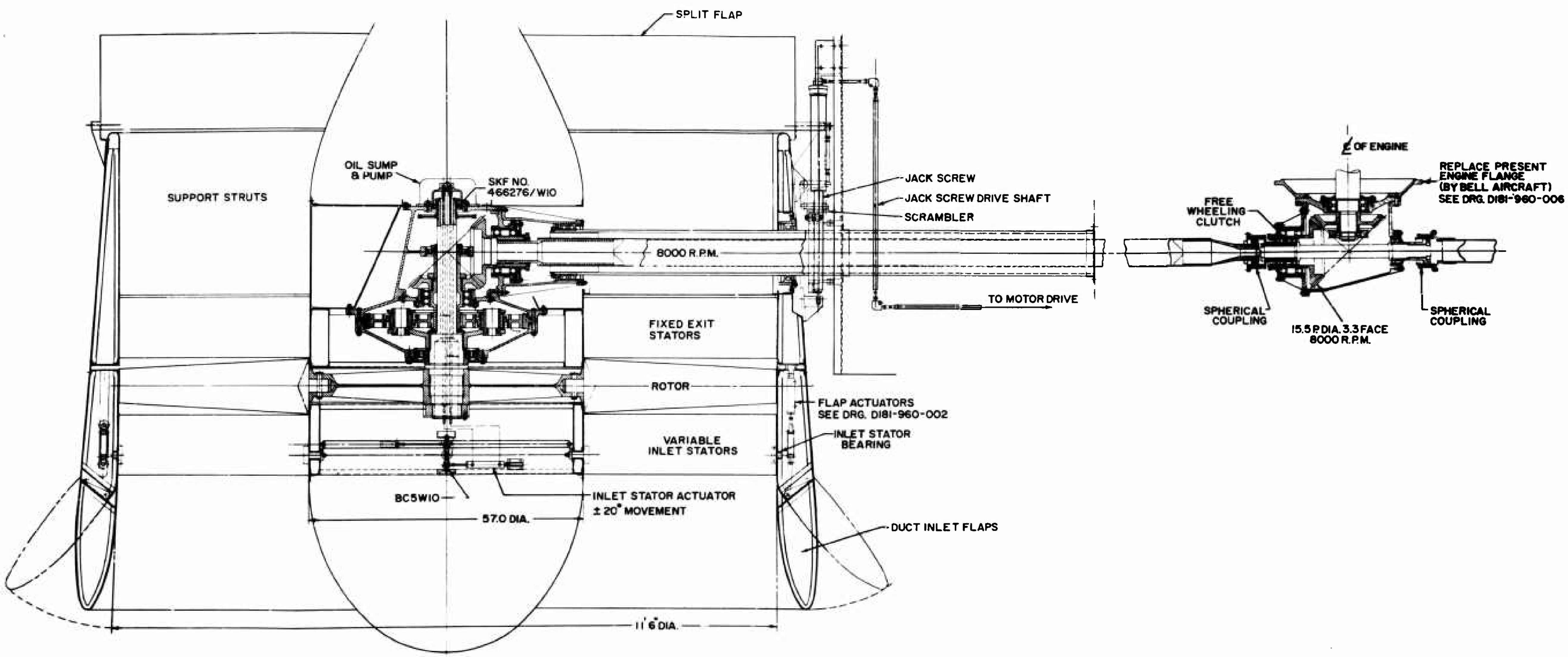


Figure 30. Propulsion System Assembly

A third method was investigated using the simple flapping of the duct leading edge ahead of the propeller. This has proved to be the least complex and most practical of the methods investigated, and will be used if the aerodynamic performance of such an inlet is acceptable. The flap segments are simultaneously pushed out by means of several actuators located within the duct profile. A positive locking system is used to hold the flaps closed in event of power loss.

The practical variable inlet duct design which has resulted from the study can also be seen in Figure 9. The duct leading edge is opened out to form a modified bell mouth inlet when the system is operating at static take-off conditions. After transition into forward flight, the flaps fold down to form the high speed low drag duct contour necessary for good forward flight performance.

Duct Rotation System

A typical system devised for rotation of the ducted propeller units is presented in Figure 31. Although it is shown for a four-duct installation, the basic element of coordination is evident. It is imperative that all the ducts move in unison so that no unusual thrust conditions will occur during duct rotational phases. A standby system will operate from the auxiliary power source in case of a primary system failure of any sort.

Powerplant Selection

In the course of design studies, turboprop engines in the 3000 hp to 10,000 hp range were considered in specific configurations. Engines that were considered included the T54, T56, T49, RB109, and Allison 550-B1 turboprops. Advanced types such as the RB109 and the Allison 550-B1, which will be available in the pre-1960 era, have proven to be very good for application to the ducted propeller transport design.

Reduction Gear Design

In general, the propeller speeds suitable for ducted propeller operation are higher than the corresponding speeds required for bare propellers. In those applications where the propeller is mounted directly to the reduction gear output shaft, the weight of the gearing will be lighter than the normal lower output speed system. For simplicity and a measure of conservatism, the reduction gear weight has been assumed equal to the basic engine gear box weight. In those instances where two engines are coupled to drive a contrarotating propeller, the basic engine reduction gear box weight is again used in the estimations of the redesigned gear box.

For those configurations where the power is shafted from the engine to separate propeller locations, it was determined that the highest practical shaft speed should be used to minimize weight. This indicates that the normal engine reduction gear box should be removed and replaced by a simple one-to-one ratio right angle spiral bevel gear set.

The development of reduction gear systems is necessary in both of the types mentioned above.

Engine Operation

The expected operational areas for the aircraft include locations at 6000 feet. altitude and 95°F temperature. Under these ambient conditions, the standard sea level ratings of turboprop engines will be reduced by about one-third. It has been found that it is possible to recover full sea level performance by injection of water into the engines (Reference 5). This has been confirmed in discussions with engine manufacturers. For this study it has been assumed that the engines will be equipped with a water injection system to recover sea level power under hot day and altitude take-off conditions. It may be mentioned here that water injection augmentation systems are standard equipment on many turbojet and turboprop engines in operation today.

The ducted propeller units have been designed to deliver take-off thrust sufficient to perform normal VTOL operations at 6000 feet

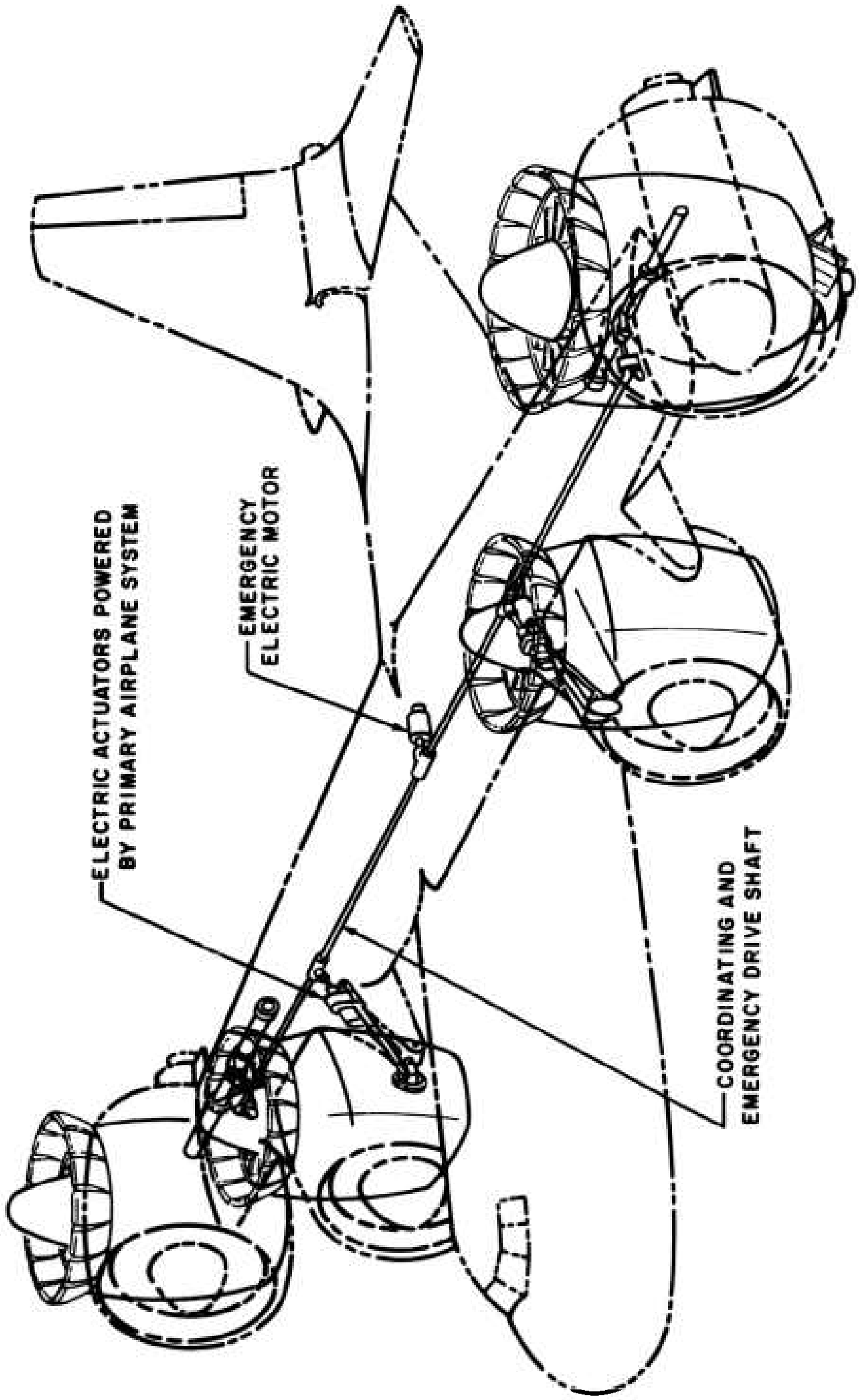
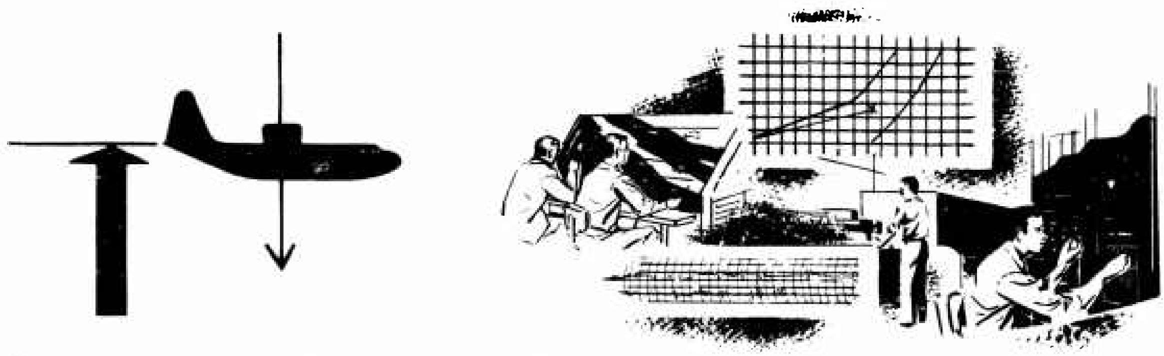


Figure 31. Duct Rotation System

and 95°F using the sea level power recovered by the water injection. Sea level thrust available from this system will be about 10 percent greater than that obtained at the 6000 feet altitude due to the increased density of the air through the ducted propellers at the sea level altitude and temperature.

The later stages of the study have shown decided advantages for the centerbody-mounted turboprop engines in those configurations known as the tilting-engine types. The advantages gained in weight and simplicity are partially offset by the additional engine development entailed. The tilting engine concept will require the engines to be operated in all positions from the normal horizontal position to forward of the vertical position or from zero to about 110 degrees. This will present additional lubrication problems which will require development of a suitable system to permit operation of the engine under these new conditions. Again it must be mentioned that several turbine engines have already been modified to operate under the same conditions as required in the present case. The problem will be somewhat relieved by the limited time of operation in the vertical and intermediate positions. The engine manufacturers have been made aware of the possible change in the mode of operation. It would be advantageous to introduce additional operational requirements early in the development of any new engine which could be applied to the selected aircraft designs, since the features could be more easily incorporated in the development stages than in the production stages of the engine.



GENERAL AERODYNAMICS STUDIES

General

The aerodynamic studies completed during this contract were divided into two parallel efforts. The first study was directed toward evaluating the propulsion systems required for a ducted propeller installation. The second phase concerned the design and evaluation of airplane configurations suitable to perform a VTOL Assault Transport mission.

In completing the first of these tasks, a comprehensive review of the available literature was considered a logical starting point. This review was followed by an extensive study of the ducted propeller from the standpoint of theoretical momentum and blade element analyses. The study resulted in the detail design of a group of ducted propellers representing at least three different approaches to the problem. The study firmly established the feasibility of the ducted propeller as a VTOL powerplant.

The design and evaluation of the airplane configurations suitable for the VTOL assault transport mission resulted in extensive analyses of the airplane performance capabilities and a study of the stability during the hovering, transition, and level flight conditions. The performance results indicated that the design could easily exceed the minimum speed requirements of the study and also demonstrated that the airplane had a wide range of mission capabilities. The stability and control analysis was principally aimed at delineating problem areas. It did demonstrate a feasible system of control and established the fact that the transport was manually controllable in hovering and transition.

These studies have been presented as four aerodynamic reports which are briefly summarized in this section.

Performance Studies

A performance analysis of ducted-propeller assault transport aircraft capable of vertical take-off and landing was made as part of the contract. Although the work was not required to comply with detailed specifications, the intent of MIL-C-5011A was followed where particular requirements for this study were not written. A generalized parametric study was completed early, in order to determine the design regime for the specific configurations. As a result, two configurations, which were similar except for the size and power loading of the ducted propellers, were selected for detailed analysis and comparison. An analysis of vertical take-off was made to establish useable and optimum thrust-to-weight ratios, and to define the performance of the particular aircraft. Analysis of the short take-off capability of these airplanes, and the resulting increased performance, was also made. The configuration (D181-960-009) powered by six Allison 550-B1 gas turbine engines had smaller ducts, higher propeller power loadings, and was concluded to be the most promising for development as a VTOL Assault Transport.

Optimum Airplane Configuration

The performance capabilities of the most promising configuration, D181-960-009, are shown in Figure 2. The classical performance indicated a high-speed potential of between 460 and 527 miles per hour and an altitude capability in excess of 50,000 feet. The airplane was powered by six Allison 550-B1 engines operating in four ducted propeller units as has been shown in the preceding sections. The engines were mounted integrally within the ducts to take advantage of the residual thrust in vertical flight. The power available is shown Figure 32, while the residual thrust is shown in Figure 33. These figures were drawn from the manufacturer's specification, (Reference 6). The propellers used were chosen as part of the propeller design study of Reference 2 and do not necessarily represent the best detail design choice. Those used in the wing tip ducts, which housed two engines each, were contrarotating, 10-bladed, variable pitch propellers. The inboard ducts housing one engine each had single rotation, 12-bladed variable pitch propellers.

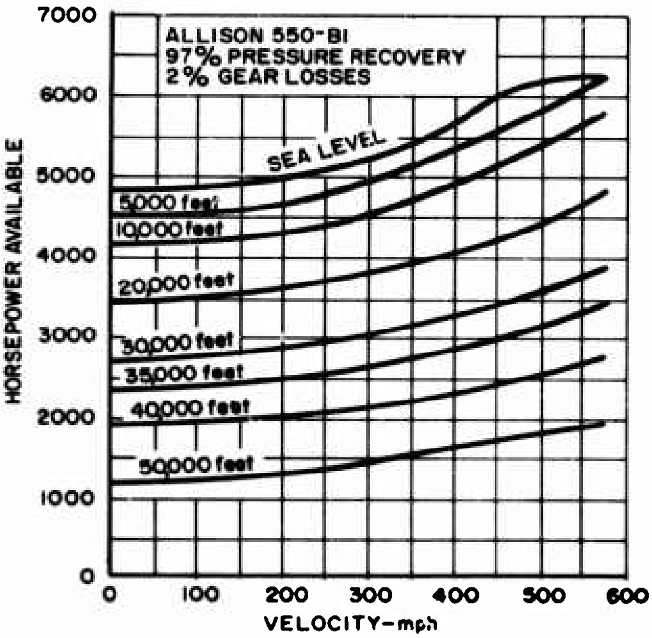


Figure 32. Power Available

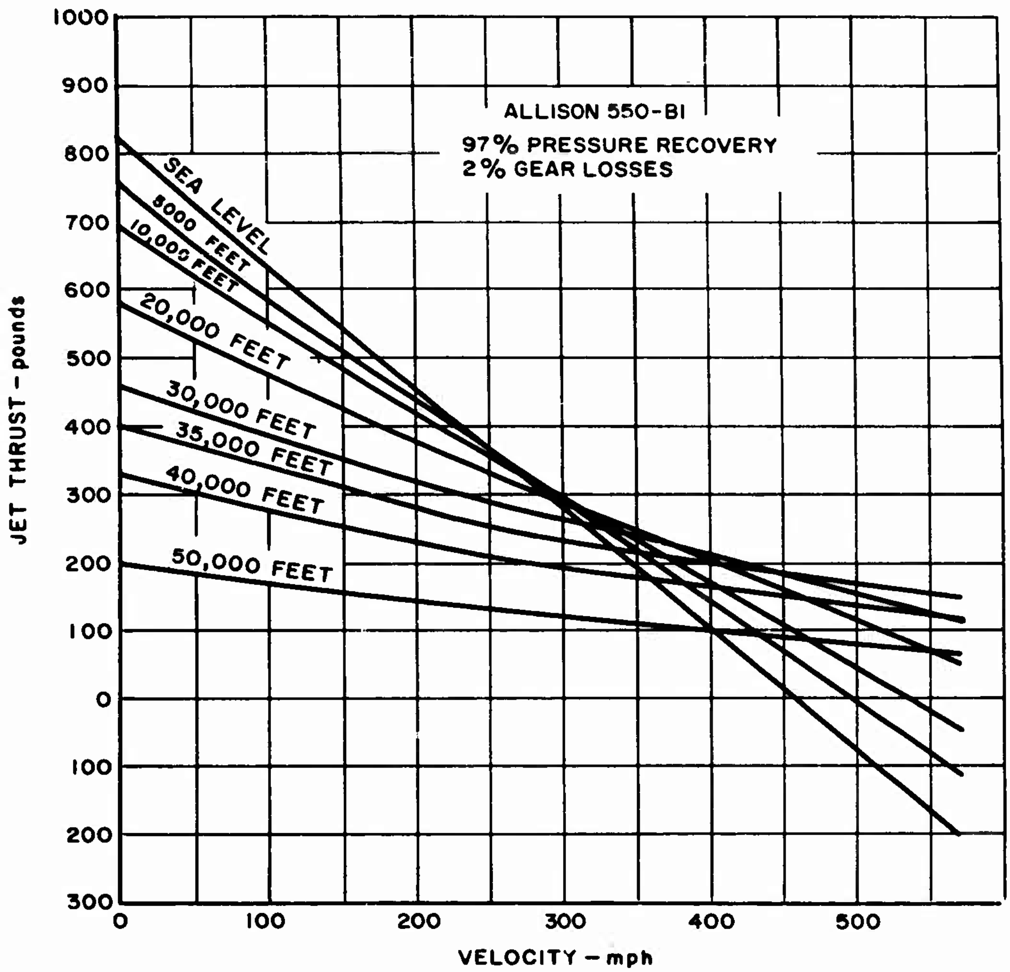


Figure 33. Residual Thrust

The thrust of these units was determined from the momentum analysis which is detailed in Reference 2. Three different detail design studies were completed for the propellers to show arrangements providing this momentum analysis thrust.

Drag Analysis

The drag analysis of this airplane was made with special emphasis on the drag of the duct units. A comparison between the drag of a high speed, and a static inlet duct was shown in Figure 11. The drag of the bell-mouth type, static inlet, required for high static thrust, was found to be about 6-1/2 times that of a high-speed duct. To avoid the drag penalty associated with the static inlet, this analysis clearly pointed to the need for variable geometry flaps on a high-speed inlet in order to achieve the take-off configuration. The drag of the high-speed inlet was appreciable, but still tolerable for the flight range. The inlet flaps were used for all ducted propeller types.

Performance

The airplane has a sea level rate of climb of 10,700 feet per minute at a gross weight of 60,000 pounds, requiring 3.6 minutes to climb to 30,000 feet, and has a service ceiling of 51,100 feet. The classical performance variations with gross weight were shown in Figure 2. The speed variation is small and even at weights up to 80,000 pounds, altitudes well above 40,000 feet are possible. The ability of this airplane to perform under emergency conditions induced by the loss of power was also examined. A representative drag and thrust curve is shown in Figure 34 at 10,000 feet altitude. The airplane was capable of flying to an altitude of 20,000 feet with four of the six engines out. This is equivalent to using only the two inboard ducts or a single wing tip duct. The rudder was fully capable of trimming the airplane under the latter condition. An investigation of the loss of power in hovering and transition flight was also completed. Loss of one engine during hovering would result in a total loss of one-third of the thrust, since failure of an engine would require automatic shut-down of the symmetrically opposite engine. The descent from 50 feet under various conditions is shown in Figure 15. With four engines operating, the impact velocity was reduced to 21.5 mph. Control would be maintained throughout the descent due to the thrust symmetry. If an engine was lost during the transition to level flight, a landing could be made if that failure occurred above the lowest equilibrium flight speed of 34.5 mph at a gross weight of 70,000 pounds, under sea level standard conditions. This condition is shown in Figure 35.

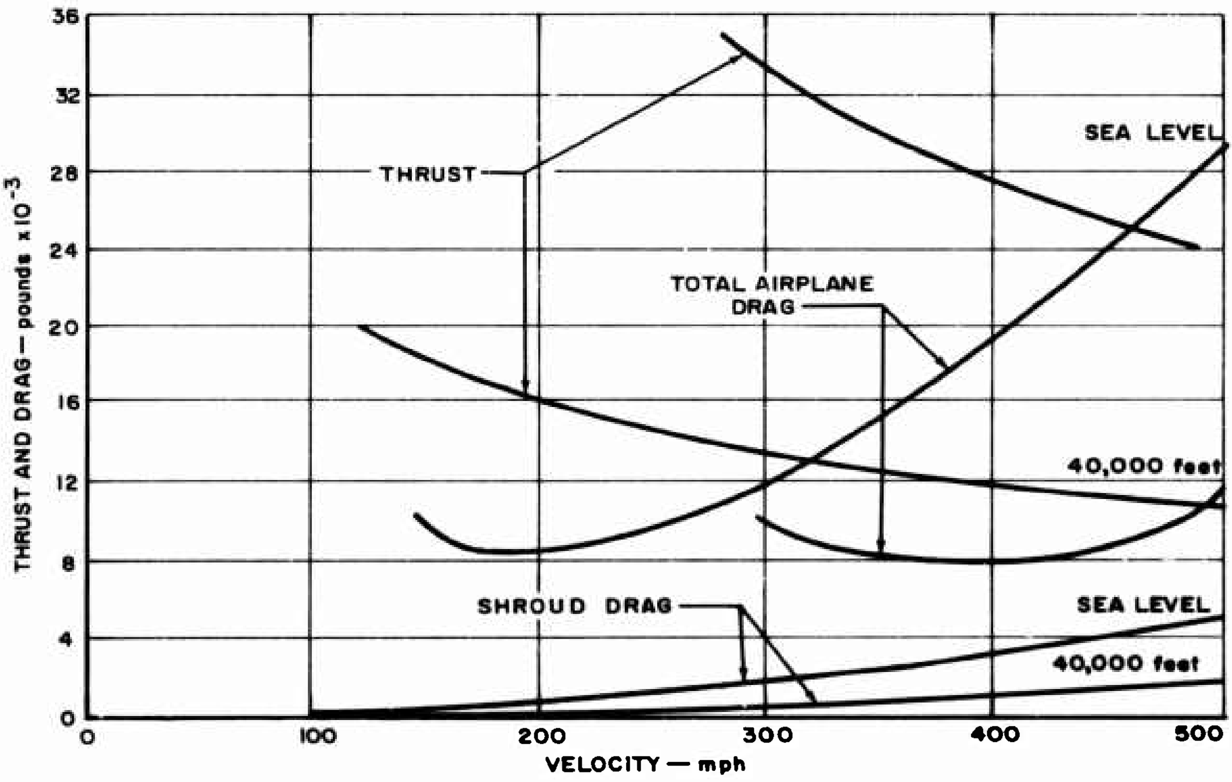


Figure 34. Thrust and Drag

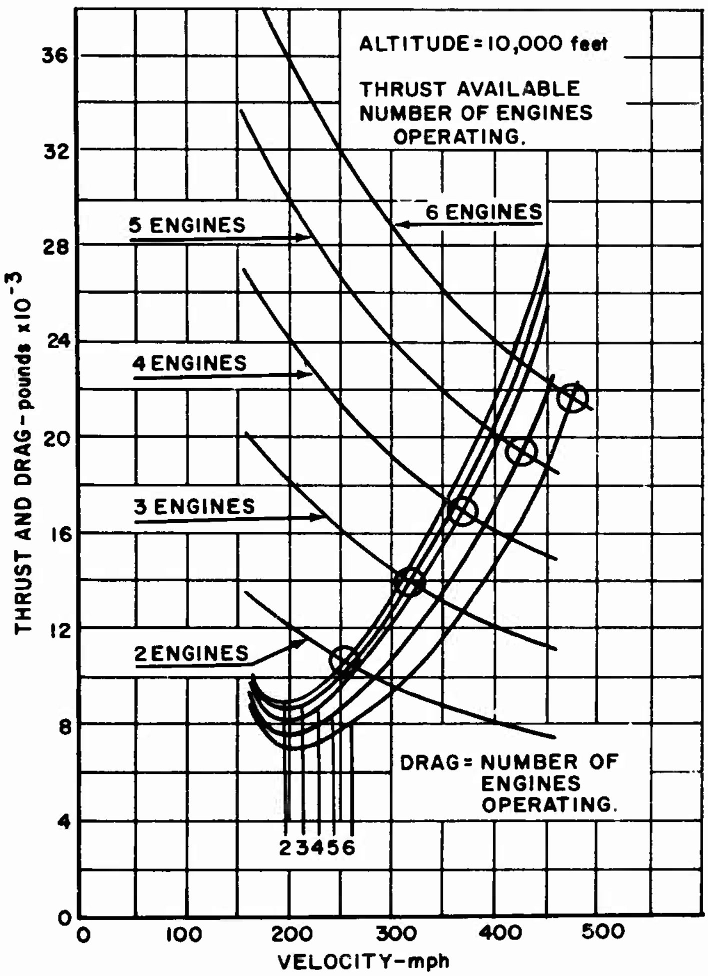


Figure 35. Emergency Operation under Reduced Power

VTO Analysis

The vertical take-off flight path used for these studi^s required only a few seconds to complete. Therefore the danger of engine failure was prevalent over a very small percent of the flight path. The VTO consisted of a vertical rise to hovering at 50 feet, during which the vertical velocity was limited to 5 feet per second and following this is a transition, during which the thrust was rotated intermittently to accelerate the airplane horizontally at low angle of attack and constant altitude to speeds for conventional level flight. The airplane power-off stall speed versus weight is shown in Figure 36. Vertical take-off calculations for the D181-960-009 are summarized in Figure 37, which shows that take-off and transition could be accomplished in about 27 seconds. The acceleration distance after the vertical rise to 50 feet was 1500 feet. The effects of a 40-knot wind were found to be beneficial. The steady wind provides an extra margin of both lift and control during the vertical rise. Figure 38 shows that

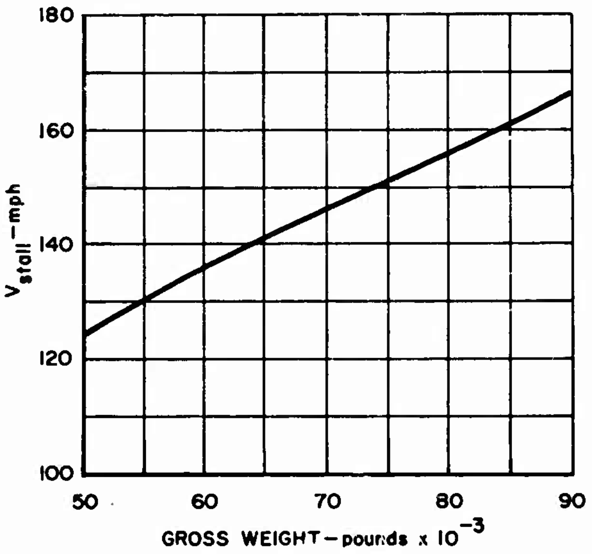


Figure 36. Power-Off Stall Speed

take-off time was reduced to 19 seconds and the air distance covered during the acceleration was 350 feet. The effects of unsteady winds were found to be controllable; and the average wind gives the same benefits as a steady wind.

The vertical landing flight path procedure started with a thrust rotation from horizontal to vertical at idle power in a normal glide. A flare to a horizontal flight path is made at the deceleration altitude (nominally chosen as 50 feet). As the airplane decelerates at constant altitude and angle of attack, the thrust is increased as the lift decreases. Rotation of the thrust about 10 degrees beyond the vertical gives a decelerating component to the thrust and shortens the deceleration time and distance. When hovering is reached, a vertical descent with a maximum vertical velocity of 5 feet per second is made. The study indicated that with a three percent thrust margin, the five-foot-per-second vertical velocity can be dissipated

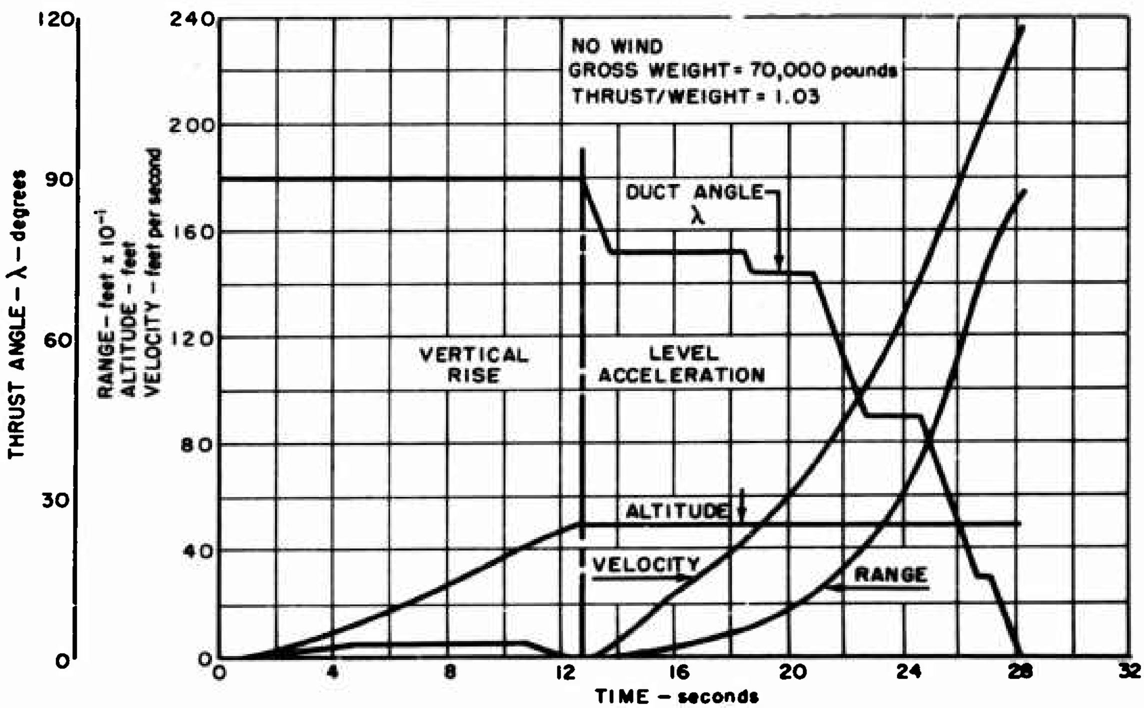


Figure 37. Vertical Take-Off and Transition

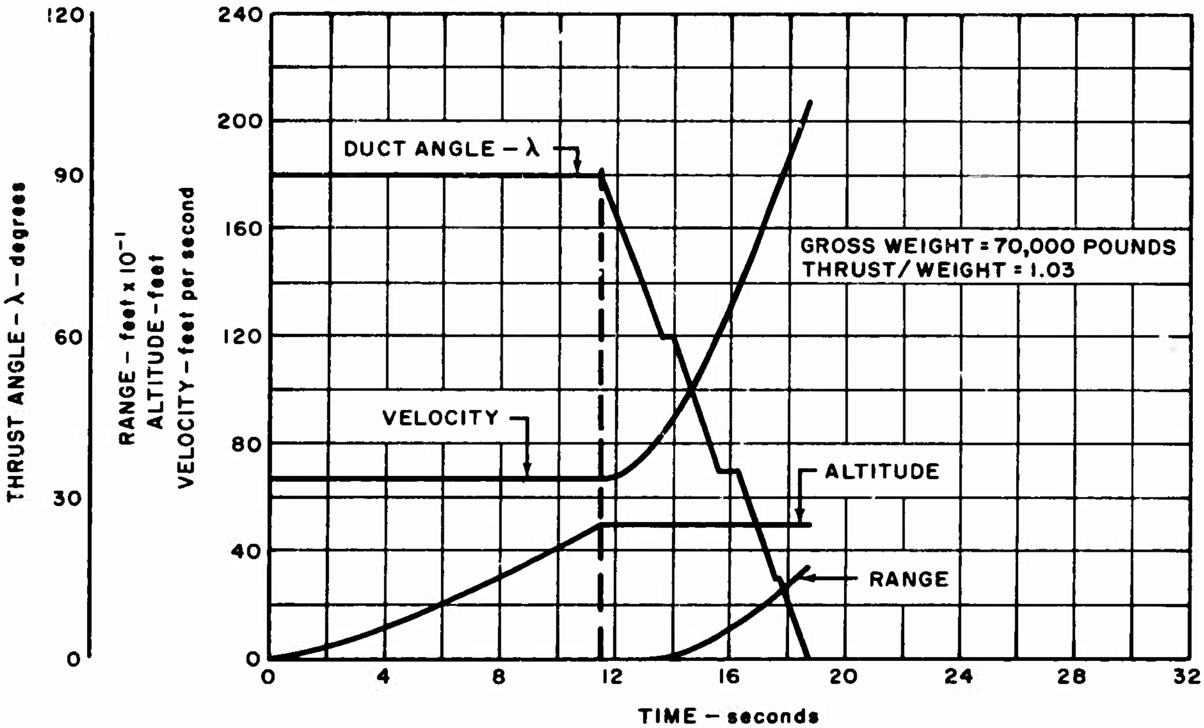


Figure 38. Vertical Take-Off and Transition in 40-Knot Wind

in about 12 feet. This led to the use of 70,000 pounds as the maximum vertical landing weight for the configuration. The landing of the airplane at a gross weight of 50,000 pounds is shown in Figure 39. The landing took 41.5 seconds and covered 2420 feet during the deceleration, which was accomplished at zero lift. The time to hovering was 29.5 seconds. At a gross weight of 70,000 pounds the total time was 45 seconds and the distance was 3120 feet. The time to hovering was 33.2 seconds.

The airplane could accomplish the assault transport mission with a minimum gross weight of 67,380 pounds. This weight allowed a fuel quantity of 13,290 pounds, which was sufficient to accomplish the radius. The maximum vertical take-off gross weight at 6000 feet and 95°F, while maintaining a 3 percent thrust margin, was 70,000 pounds. The thrust required to provide T/W = 1.03 was obtained by using water injection at these conditions. If the 70,000-pound gross weight was used during VTO, the airplane was capable of performing missions with radii and pay loads in excess of the basic requirement since additional fuel could be added. The basic mission requirement for the transport was to execute a vertical take off at 6000 feet and 95°F and fly a radius of 425 miles at 300 miles per hour. Twenty percent of the distance was to be flown at sea level, while the remainder was at best altitude. The pay load was to be 8000 pounds out and 4000 pounds back with a vertical landing and take-off at the radius point and no fuel addition. A ten percent fuel reserve was held. Figure 24 showed the basic mission could be accomplished with the take-off gross weight of 67,380 pounds and an average cruise altitude of 27,000 feet. In addition, the use of sea level standard VTOL conditions pro-

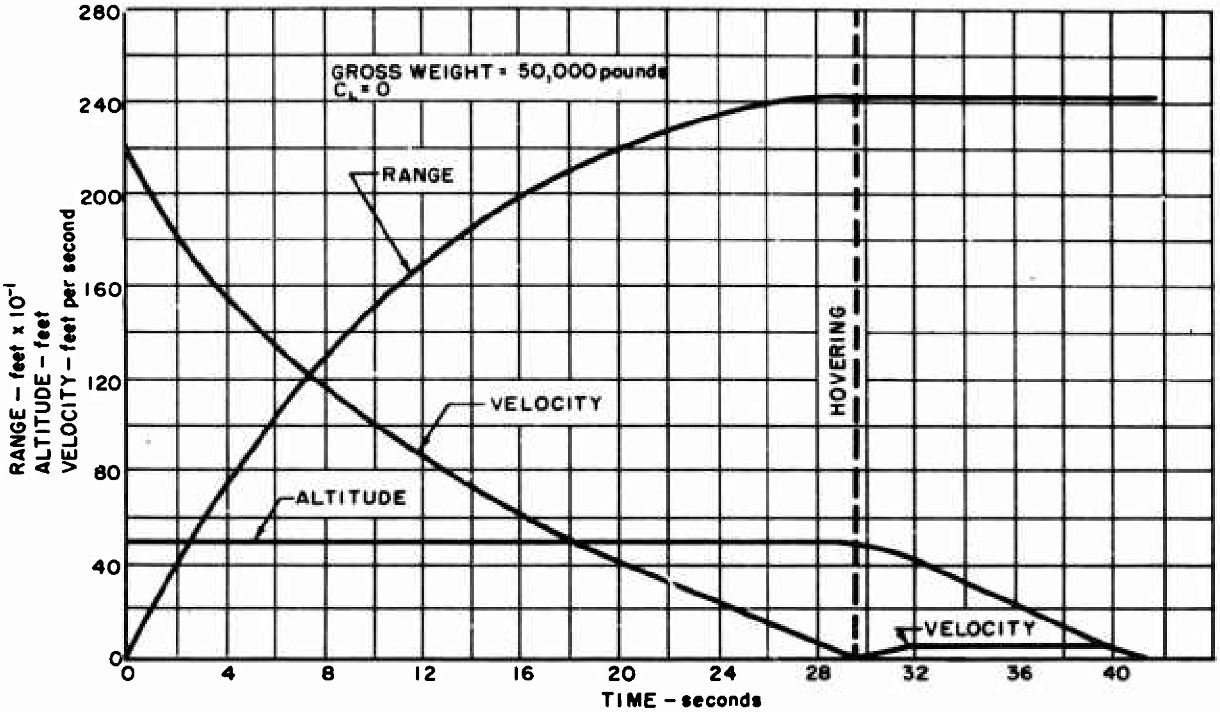


Figure 39. Landing Performance

vided a 705 mile radius. Alternate STO capability provides a maximum mission radius of 987 miles. Various extensions and modifications were possible utilizing the vertical take-off capability at 70,000 pounds and the additional fuel load this permitted. Using the 70,000-pound VTO weight, the complete basic mission could be accomplished at a minimum cruise altitude as low as 11,300 feet. Maintaining the cruise altitude of 27,000 feet for the 80 percent segment, and using a 20% sea level portion, this radius could be increased to 513 miles with this additional fuel. If both radius and altitude were maintained while using the 70,000 pound take-off, the pay load could be increased to 10,500 pounds. A mixed speed mission showed that the speed at altitude could be increased to 420 mph in conjunction with 300 mph at sea level to carry the 8000 pound pay load 425 miles. A high-speed mission was investigated by using a cruise speed of 455 mph during both the altitude and the sea level segments of the radius. This represented a 50% increase in speed over the minimum specified, yet even at this high speed the airplane was capable of a 302 mile radius. This high-speed radius is actually a very useful operational distance. VTOL gross weight of 75,800 pounds was possible with standard conditions at sea level and, as noted previously, a 705-mile radius resulted. Some of these missions are summarized in Figure 5. The upper portion of the chart shows VTOL missions while STO - VTOL missions are shown in the lower section.

STO Missions

This airplane combines the best features of an STOL airplane with its normal VTOL characteristics. Several alternate mission profiles, which used an initial short running take-off, with vertical landings and take-offs at all other points were investigated. The overload was limited to that which would allow a

70,000-pound gross weight vertical landing at 6000 feet and 95°F at the advanced base. Under these conditions, the 425-mile radius with the basic 8000-pound pay load could be increased 2.3 times to 987 miles by overloading to a gross weight of 86,150 pounds. This overload required a take-off ground roll of 770 feet. In the event a radius of 425 miles was adequate, an increase in pay load on the outbound leg from 8000 pounds to 16,720 pounds could be made. This increased the take-off gross weight to 76,890 pounds and required a 300 foot take-off ground roll. A high-speed mission has shown a radius of 607 miles to be possible with a 450 mph cruise velocity with a take-off gross weight of 83,530 pounds using a 660-foot ground roll. On these missions, the heavy weight, vertical landings were made at a weight of 70,000 pounds, which allows a 3 percent thrust margin at 6000 feet and 95°F. These missions define specific points of radius potential and indicate the versatility of the vertical take-off transport designed with a short take-off capability.

The ferry range with an 8000 pound pay load was investigated with an initial vertical take-off, and with a STO of 400 feet and 800 feet ground roll (Figure 40). With a vertical take-off at 6000 feet and 95°F at a gross weight of 70,000 pounds, the ferry range was 1360 miles. With a 400-foot ground roll, at gross weight of 78,460 pounds, the range was 1850 miles. The range with an 800 foot-ground roll, at a gross weight of 86,760 pounds, was 2520 miles. These ranges were accomplished at 30,000 feet at a cruise velocity of 320 mph. If the quantity of fuel was increased to replace the 8000-pound pay load, a maximum range of 3260 miles was possible.

In addition to the radius capabilities of the airplane, it is interesting to note the flexibility represented by the hovering capability. Figure 7 showed the hovering time available at various ranges, assuming a 6000-foot, 95°F day condition for various VTOL and STO conditions. It is also possible to increase these numbers from the maximum of 74 minutes

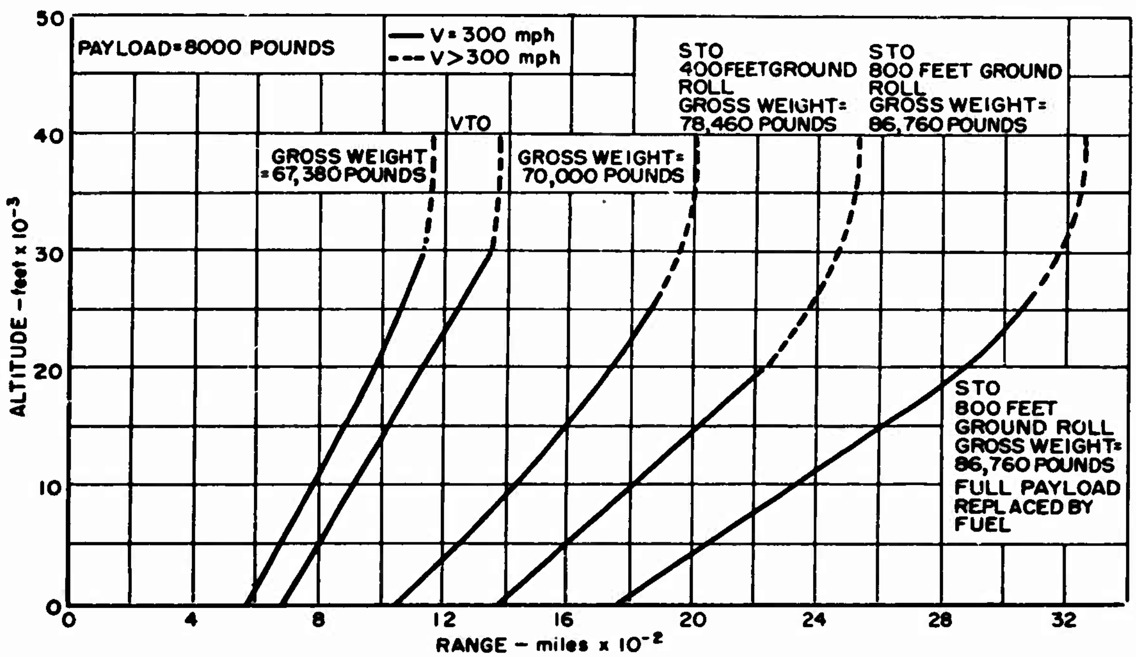


Figure 40. Ferry Range vs Cruise Altitude

shown to 100 minutes, if sea level standard VTOL is considered. The minimum hovering time represents 10 percent of initial fuel in each case.

If a short runway is available, this aircraft can take advantage of the runway to execute a rolling take off with an overlaod of fuel or payload, thereby tremendously increasing its potential. Calculations were made to determine the take-off distance required for overload conditions of thrust less than the weight. To perform the rolling take off, the ducts are rotated to some predetermined position between the horizontal and vertical, resulting in the ground roll distances shown in Figure 41. A typical thrust-weight ratio of 0.88, which is a gross weight of 82,000 pounds, shows an angle for minimum ground roll of 50° from the horizontal, resulting in a ground roll distance of 584 feet. With the duct in this position, the airplane is accelerated to lift-off speed. After lift-off, the climb to 50 feet is made without further rotation of the thrust. When 50 feet is reached, the thrust is rotated to the horizontal as the airplane accelerates to level flight speed. The optimum ground roll characteristics summarized in Figure 4 shows the overall VTO-STO capability of the airplane in terms of total take-off distance at various gross weight conditions. At all weights up to 72,000 pounds VTO is possible, resulting in zero distance. As the weight is increased, a small ground roll distance is required. At 80,000 pounds, 500 feet of ground roll and 1180 feet total distance are required. It is quite significant to note how large the transition distance is for small overlaods in weight, demonstrating only a small gain in pay load in switching from pure VTOL to STO of 500 feet over 50-foot obstacles. This seems to indicate small design relief in moving from a VTO to an STO requirement.

A detailed weight statement for the -009 configuration is presented in Table VII.

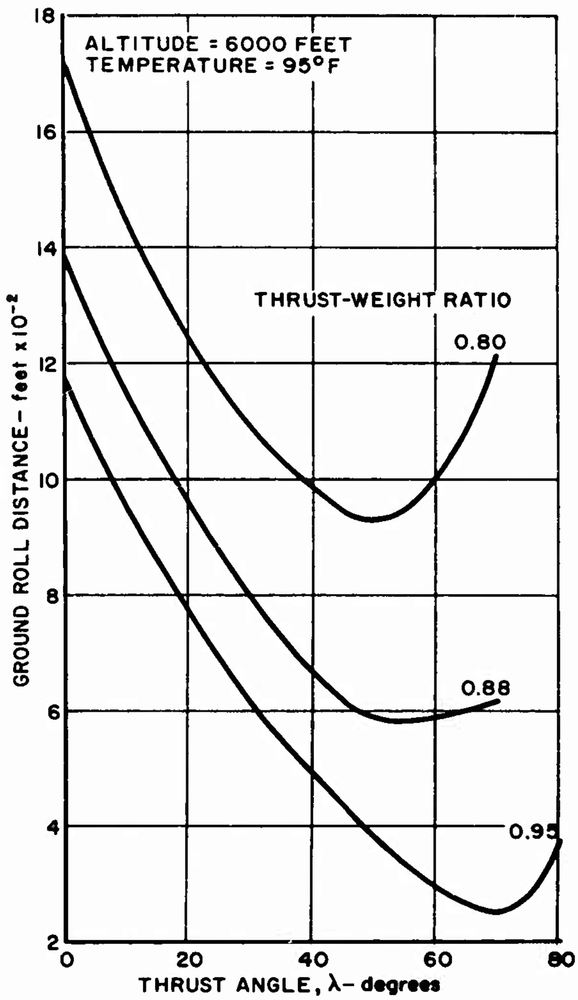


Figure 41. Short Take-Off Ground Roll vs Thrust Angle

TABLE VII. FOUR DUCT ALLISON 550-B1 TILTING ENGINE CONFIGURATION

D181-960-009

		Weight pounds
Wing		5,200
Tail		
Horizontal		685
Vertical		486
Body		7,423
Landing Gear		300
Surface Controls		1,930
Flight Controls		500
Reaction Controls (Pitch)		400
Engine Section (Duct Around Propeller)		
Inboard (Vertical Position)		2,780
Outboard (Vertical Position)		3,920
Propulsion		
Engines (2) Inboard - Allison 550-B1		3,150
Engines (4) Outboard		6,300
Inboard Gear Boxes (2)		980
Outboard Gear Boxes (2)		2,180
Engine Mounts Inboard		125
Duct Supports Outboard		245
Inboard		300
Outboard		500
Rotating Mechanism		
Inboard		60
Outboard		100
Lubricating System		195
Fuel System		460
Water Injection System		200
Engine Controls		50
Starting System		150
Propeller Installation		
Inboard (Vertical Position)		1,094
Outboard (Vertical Position)		1,522
Auxiliary Power Plant		80
Instruments		160
Hydraulics (Brakes and Nose Steering)		50
Electrical		800
Electronics		500
Furnishings (No paratroop seats)		465
Air Conditioning and Anti-Ice		500
Auxiliary Gear (Jacking, Towing)		25
	Total Weight Empty	43,815
Useful Load		
Crew (3)		645
Oil-Engines 25 Gal.		188
Gear Boxes		140
Fuel		13,295
Water		1,297
Pay load		8,000
	Total Useful Load	23,565
Gross Weight (VTOL Position)		
Total Weight Empty		43,815
Useful Load		23,565
	Gross Weight	67,380

Configuration D181-960-007

The second configuration (D181-960-007) examined in detail was powered by 6 Rolls Royce RB109 engines. This configuration was designed with lower power loading, and consequently with larger propellers than was the -009. The lower power level and higher static thrust-to-horsepower ratio of this configuration, resulted in a lower performance potential. The configuration was designed to see if any advantages in operating economy could be obtained by using lighter disk loadings on the propellers. The classical performance at various weights is shown in Figure 42. The high speed was 390 mph. At a gross weight of 60,000 pounds, the airplane had a sea level rate of climb of 7,450 feet per minute, a time to climb to 20,000 feet of 3.5 minutes, and to 30,000 feet of 6.5 minutes. The service ceiling was 42,000 feet. The mission analysis indicated that this airplane could complete the basic mission at a take-off gross weight of 70,000 pounds. The airplane used less fuel than the high performance configuration -009 but had a higher basic mission gross weight due to the larger duct sizes required to produce the same thrust with less power. The ability of this airplane to operate with an initial short take-off was also investigated. With a take-off gross weight of 82,690 pounds and an initial take-off run of 610 feet, the airplane could accomplish a radius of 831 miles. For a 425 mile radius it could carry a payload of 14,200 pounds with a take-off gross weight of 76,530 pounds and an initial ground run of 280 feet.

The ferry range with vertical take-off, and with a 400 foot and an 800 foot running take-off, was determined and is compared with the optimum configuration in Figure 43. The range

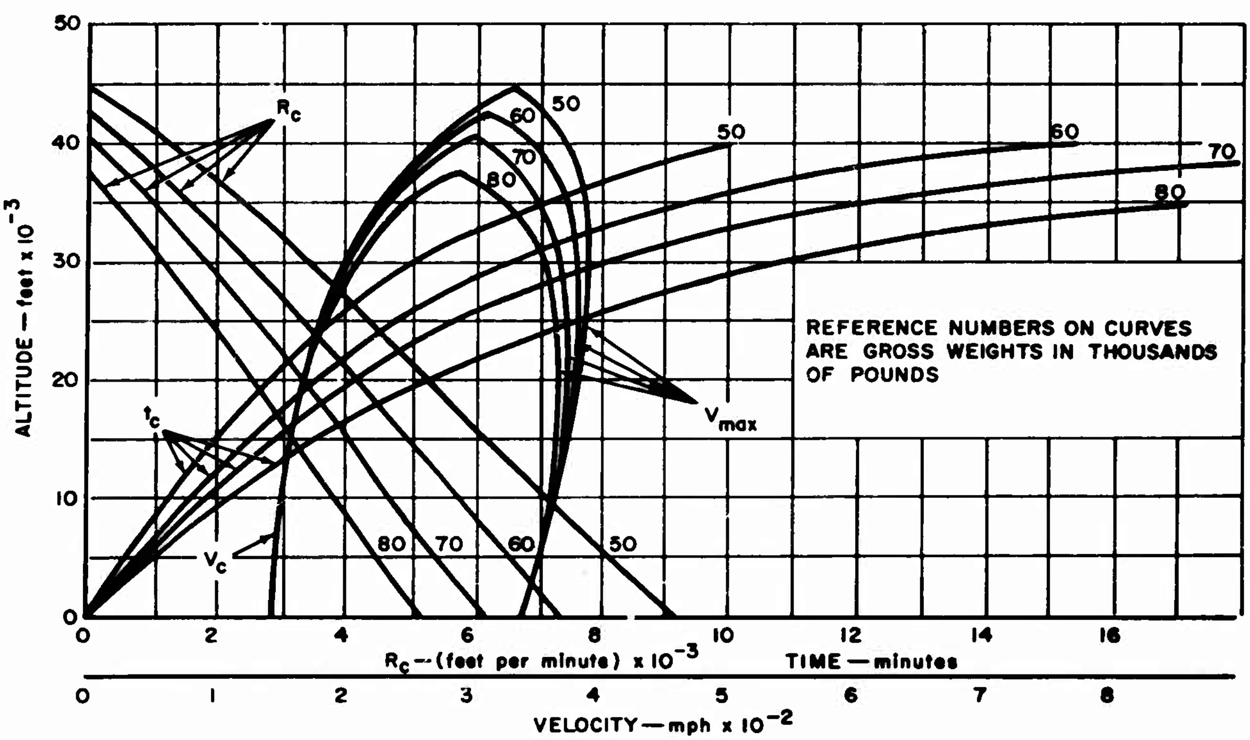


Figure 42. Classical Performance – D181-950-007 Configuration

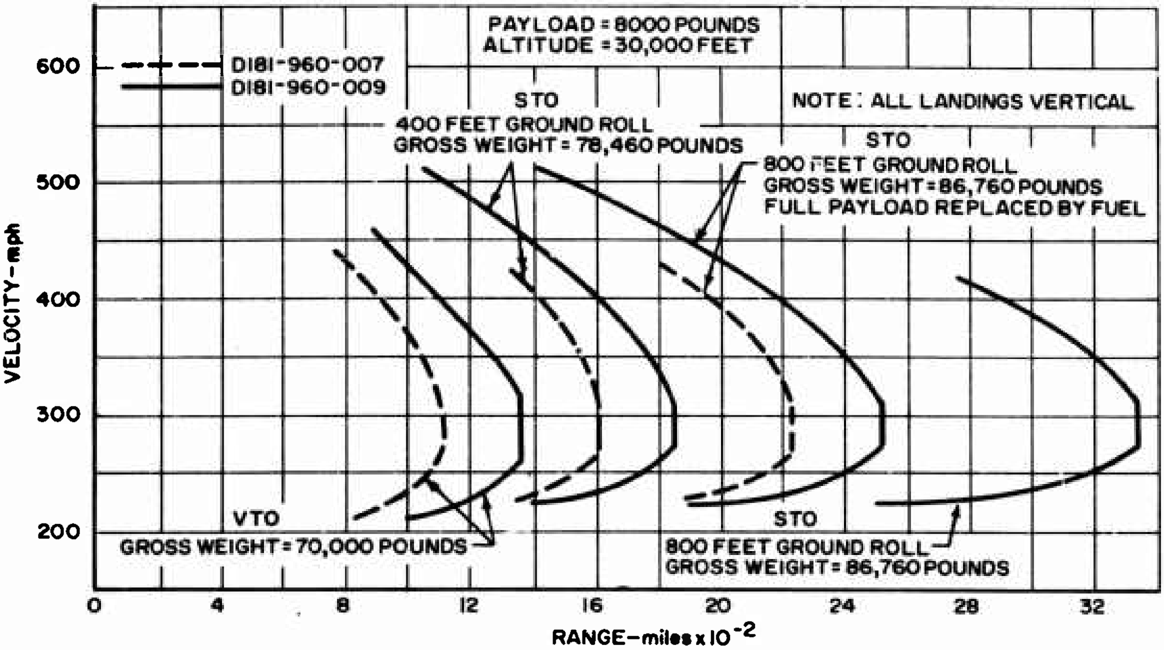


Figure 43. Ferry Range vs Speed

with a vertical take-off at a gross weight of 70,000 pounds was 1115 miles; with a 400 foot ground run the range was 1605 miles at a gross weight of 78,460 pounds. A range of 2230 miles was possible after an 800 foot take-off at a gross weight of 86,760 pounds. The cruise was at 315 mph at 30,000 feet. Although this configuration had better economy in cruise, the increase in weight due to the larger duct sizes offset this advantage and made the -009 a superior configuration.

Parametric Study, Configuration D181-960-001

The preceding designs were selected as a result of a generalized parametric analysis which was made to determine a range of aspect ratio and wing loading suitable for this ducted propeller, VTOL, assault transport aircraft. The Study configuration used, as described in Section IV, was characterized by ducts mounted at the wing tips and by booms housing the engines and supporting the empennage. The wing and booms were mounted high on a pod-like fuselage. The power from two Wright T49 engines was shafted through the wing to ducted propellers. The aspect ratio was varied from 4 to 10, and the wing loading from 30 to 60 pounds per square foot, to evaluate the influence of these parameters. The basic assault transport mission was used to evaluate the results of this study. In addition, a general analysis of vertical take-off was made for this configuration.

The aspect ratio variation indicated an increased performance advantage existed, with increased aspect ratio; that is, the fuel to perform the basic mission decreased, as shown in Figure 44, and the ceilings and rates of climb increased. However, these gains were somewhat counteracted in that the wing weight increased. When fuel saving is offset by an increase in wing weight, a region of minimum weight results which extends from about aspect ratio 5 to 7 as shown in Figure 45. The other advantages of high aspect ratio led to a choice of the range of aspect ratio between 6 and 7.

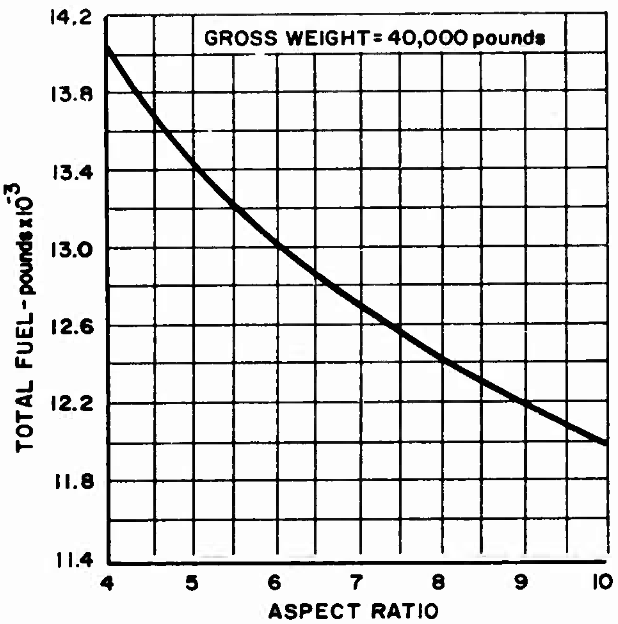


Figure 44. Total Fuel Required to Complete Basic Mission

The results of the wing loading variation study have shown that, with constant weight or variable wing area, the minimum weight of fuel plus wing occurred at a wing loading of 53 pounds per square foot. With constant wing area, the fuel to complete the mission decreased and the miles per pound factor increased as the wing loading or gross weight decreased. This trend is shown in Figure 46. These effects combined to make the best wing loading slightly lower than that indicated from wing area variation alone, and led to the choice of a range of wing loading from 40 to 60 pounds per square foot. Since the wing weight per unit area will vary from one design to another, the weight cannot be tied into a parametric study with great accuracy. The results of the study led to the choice of a range of variables rather than to specific values. The range of aspect ratio from 6 to 7 and of wing loading from 40 to 60 pounds per square foot was incorporated into the later designs.

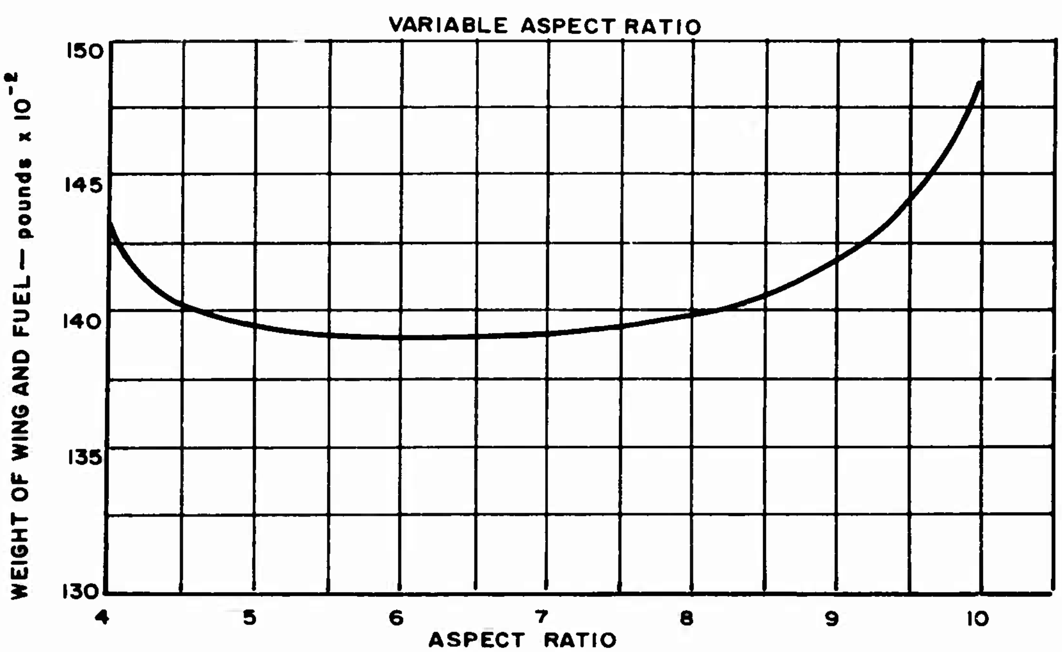


Figure 45. Weight of Wing and Fuel for Basic Mission

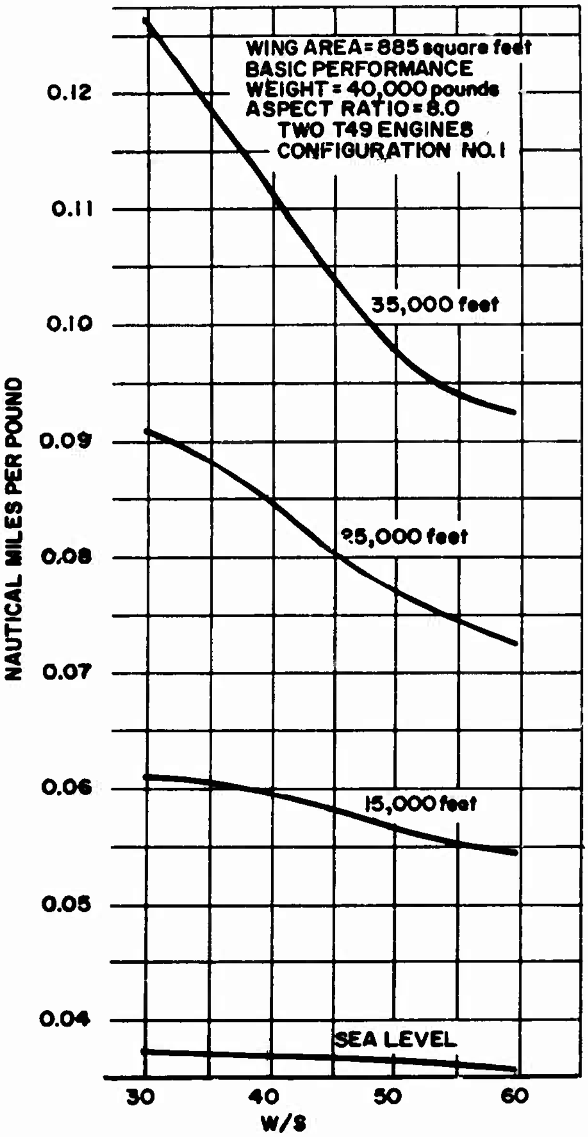


Figure 46. Cruise Parameter - Variable Gross Weight

Vertical Take-Off and Landing

An analysis of vertical take-off capability as a function of thrust to weight ratio was accomplished. Calculations of the total fuel required to complete the vertical take-off and transition to level flight as a function of initial thrust to weight ratio were made for the generalized study configuration. The time and fuel required to rise vertically to 50 feet as a function of thrust to weight ratio was also determined and is shown in Figure 47. The time to rise decreases rapidly with increasing thrust, so for a thrust increment of 3 percent, the time to rise was reduced from 51 to 13.5 seconds. At the same time, the fuel required for the take-off decreased. If this thrust increment was achieved by a reduction in weight, this would mean a loss of 1500 pounds with an available thrust of 50,000 pounds. If all this weight was fuel, the utilization of a 3 percent thrust margin would result in a net fuel reduction of 1300 pounds by the time the plane was airborn. This example shows that the most fuel, or pay load, can be taken aloft from an initial thrust weight ratio of one. This finding corresponds with previous studies on jet aircraft described in

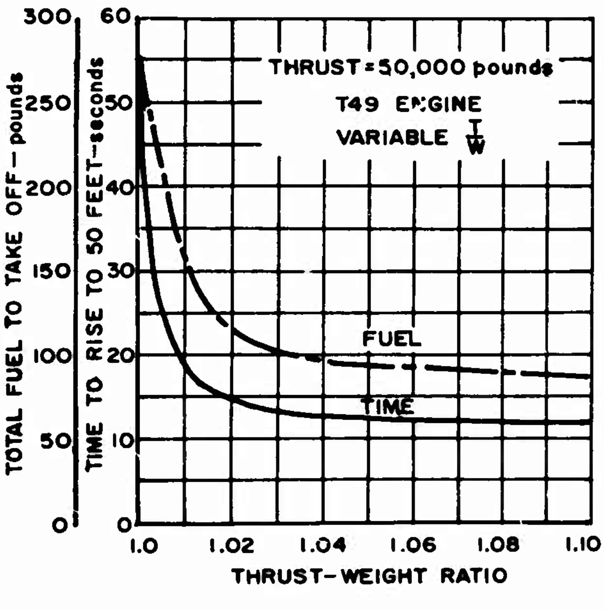


Figure 47. Vertical Take-Off Performance

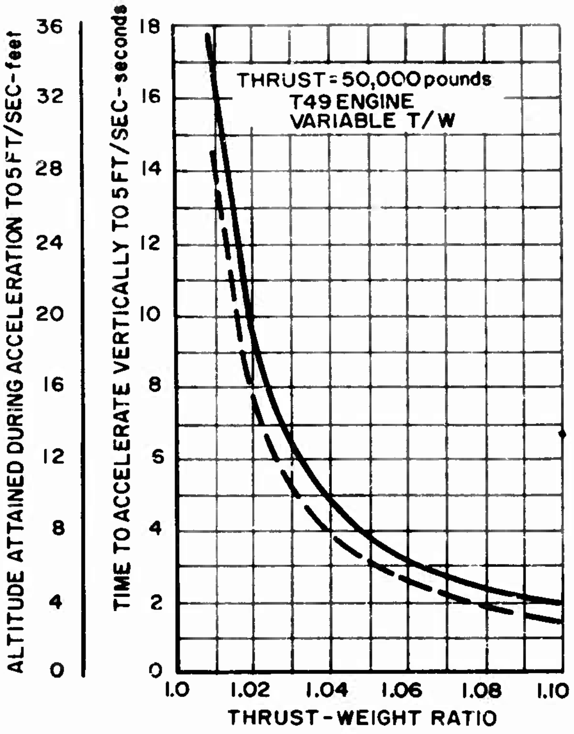
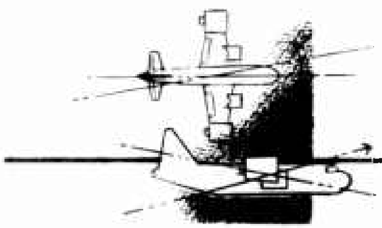


Figure 48. Vertical Take-Off Performance

References 7 - 10. However, the choice of initial thrust to weight ratio is also influenced by the time required to accelerate vertically to 5 feet per second, and the altitude attained at the end of the acceleration. Figure 48 shows that a thrust initially equal to the weight causes the vertical velocity to remain below 5 feet per second until the 50 feet altitude is exceeded. This demonstrates that $\frac{T}{W} = 1.0$ is not feasible for practical application to ducted propellers since the acceleration is too low. Using a 3 percent thrust margin, the rise velocity is reached in 5 seconds at an altitude of 12 feet. In order to obtain a positive lift-off and acceleration in vertical flight, this analysis indicates that a 3 percent thrust margin at take-off for a ducted propeller VTOL transport is desirable.



Stability and Control Studies

A preliminary analysis of the dynamics of hovering and transition flight has been completed for the ducted-propeller powered, VTOL Assault Transport Aircraft. Emphasis was placed on the control hovering studies since these investigations established the reaction control design criteria. Figure 23 shows the arrangement of a cockpit and controls to let the pilot impose control moments on a REAC machine representing characteristics of the assault transport. These studies were an extension of previous studies described in References 11 - 14. The generalized computer circuit, providing pilot control, airplane characteristics, interconnection of gyroscopic forces and random gust inputs is shown in Figure 19. Associated stability and control studies were also conducted for the transition phase between hovering and level flight as an airplane. Analogue simulation of the aircraft dynamics along with this technique incorporating conventional airplane controls enabled Bell test pilots to give "pilot opinion" concerning satisfactory and unsatisfactory flying characteristics of various reaction control input levels. True history traces of pilot and aircraft response were recorded and briefly analyzed to determine the feasibility of stabilizing hovering and to establish control design requirements. Figures 18a and 18b show the results of a study to evaluate control gradients, Figures 49a and 49b the influence of weight changes and Figures 50a and 50b the effects of gusts in pitch, roll, and yaw. Figures 51a and 51b demonstrate the fact that large aerodynamic duct influences, further improves the hovering condition.

From the hovering studies it was learned that the pilot was able to control stabilize the airplane under still air and severe gust conditions up to 50 feet per second in roll, pitch, and yaw. It was established that the aerodynamic duct influences tend to reduce ground drift tendencies and further stabilize the configuration. The engine gyroscopic coupling, as shown in Figures 50a and 50b, was not of a large magnitude due to the high thrust-horsepower ratio of the ducted propellers which permit the use of engines with smaller angular momenta than a jet. The control requirements were established as shown in Figure 21 and the following gradients were determined to be pilot acceptable:

(1) Roll reaction control 7.5 degrees/second^2

(2) Pitch reaction control 10 degrees/second^2

(3) Yaw reaction control 5 degrees/second2

The equivalent forces and moments required to obtain these gradients are also shown in Figure 21.

Preliminary analyses of the duct exit flap indicated that roll and yaw control, can be obtained by using split flaps with flap chords of one to three feet in length as shown in Figures 52 and 53. For pitch control, Figure 54 shows the downward force exerted by a J85 engine, located at the vicinity of the vertical and horizontal tail junction, will provide sufficient control for pitch attitude. The J85 reaction engine is capable of producing 2400 pounds of force for pitch control.

The stability characteristics of the airplane, during the hovering to level flight transition cases, were examined for representative speeds on take-off and landing. Three representative gust conditions representing a 50-foot-second gust in roll, pitch, and yaw are shown in Figures 55a, 55b, 56a, 56b, 57a, and 57b. Aerodynamic control from elevated rudder became more apparent to the pilot with the higher forward velocities, and added to the control already available from the hovering controls. These traces show the pilot's ability to avoid divergent motion for the various conditions investigated and emphasize the ease of controlling the transition.

The level flight stability of this airplane was quite conventional after stall speed was exceeded. The tail surfaces were designed to give an average longitudinal static margin of 12 percent as shown in Figure 58, and a yaw stability level of approximately 0.0015/degree, as presented in Figure 59. The wing geometric dihedral is zero, with positive roll stability as exhibited in Figure 60.

The following conclusions may be drawn from these studies:

1. Acceptable control gradients can be obtained for the assault transport by using available aerodynamic and reaction controls.

2. Pilots can control the airplane manually in hovering flight with the acceptable control gradients established.

3. The pitch reaction control can be obtained from a J85 engine situated at the tail end of the fuselage.

4. A split flap type control arrangement at the dust exist will provide a satisfactory amount of force for roll control.

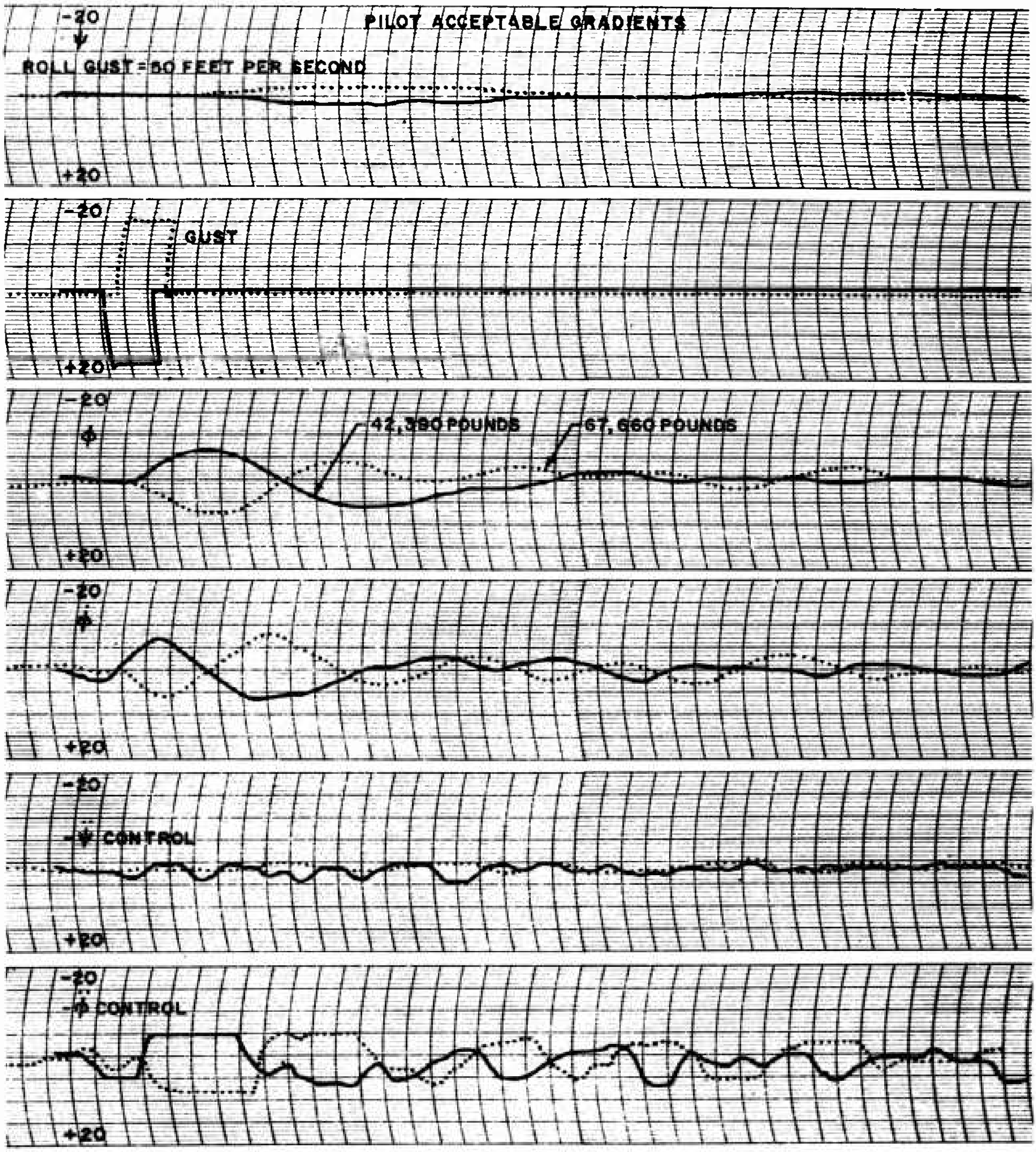


Figure 49a. Effect of Landing and Take-Off Weight in Hovering. Lateral Dynamics

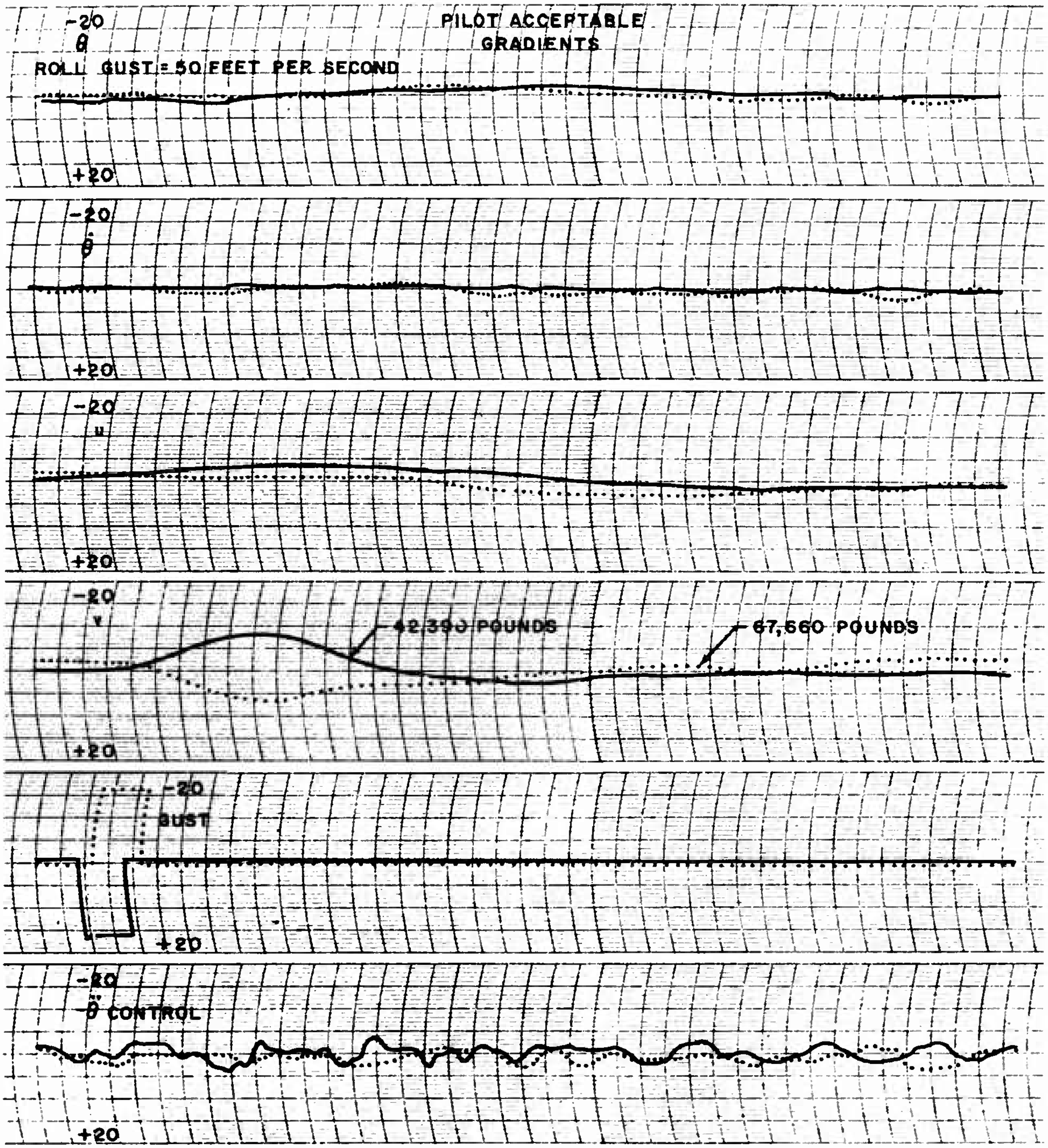


Figure 49b. Effect of Landing and Take-Off Weight in Hovering.
Longitudinal Dynamics

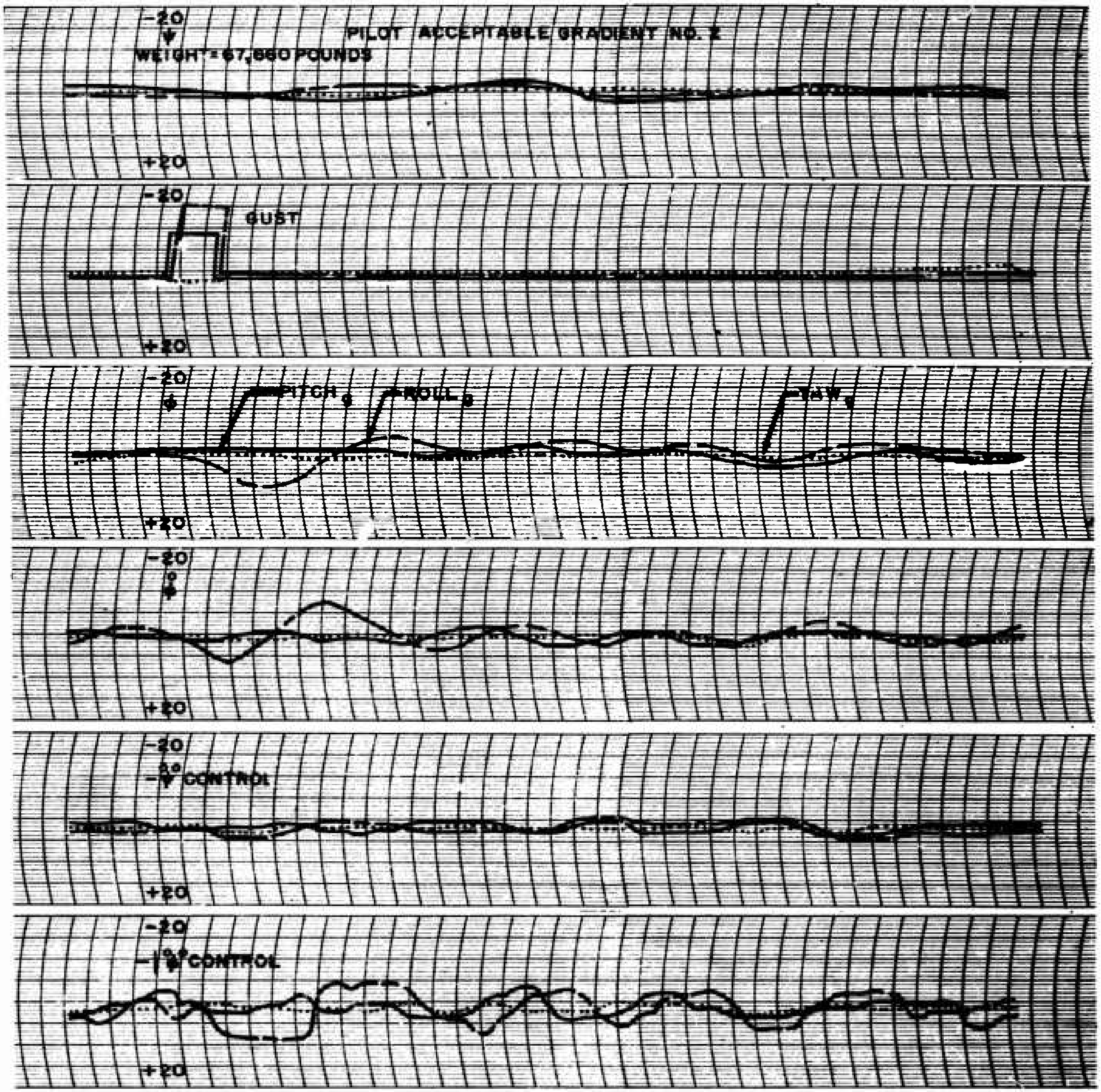


Figure 50a. Effect of Roll, Pitch, and Yaw Gusts at 50 feet per Second in Hovering. Lateral Dynamics

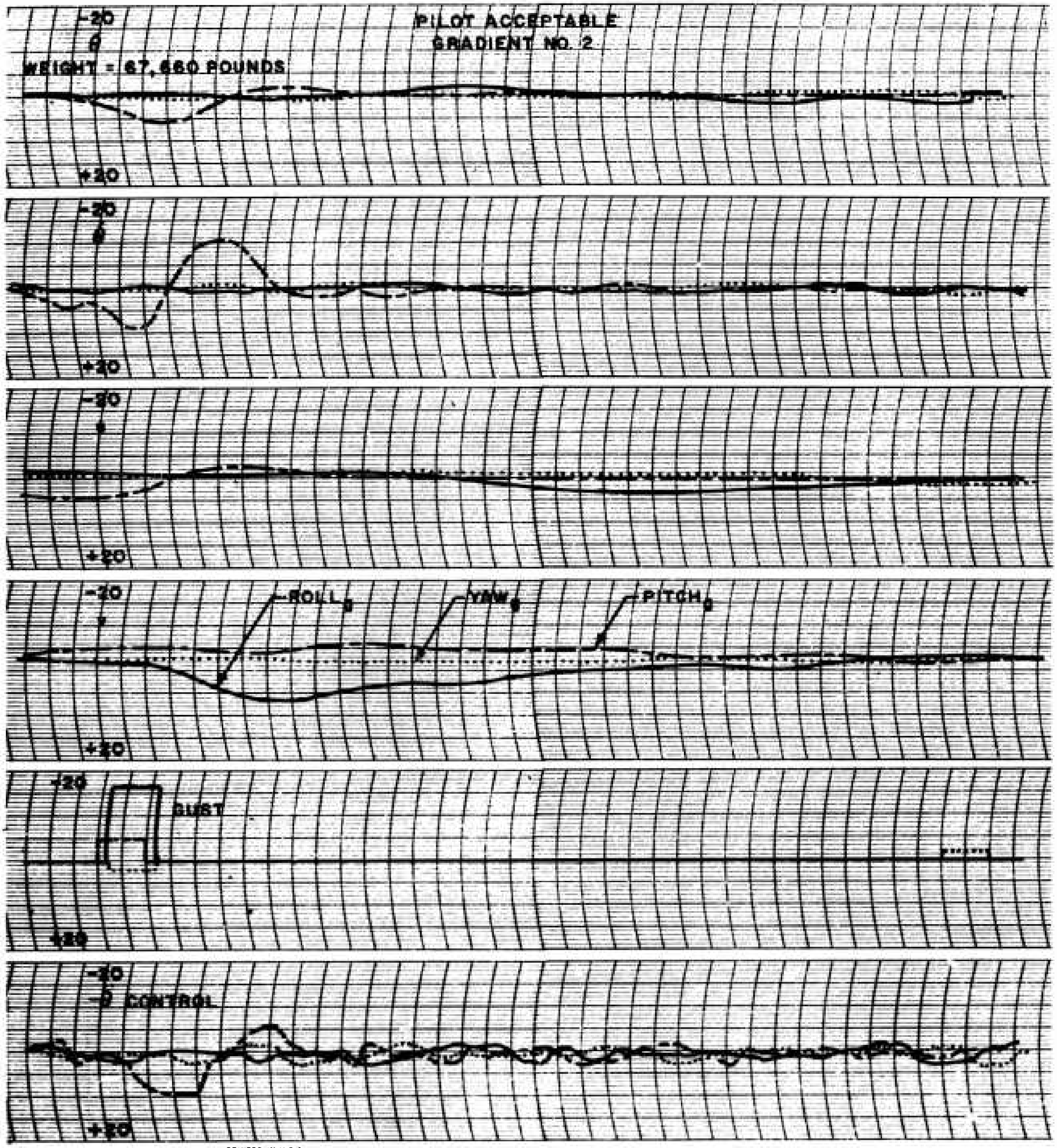


Figure 50b. Effect of Roll, Pitch, and Yaw Gusts at 50 feet per Second in Hovering. Longitudinal Dynamics

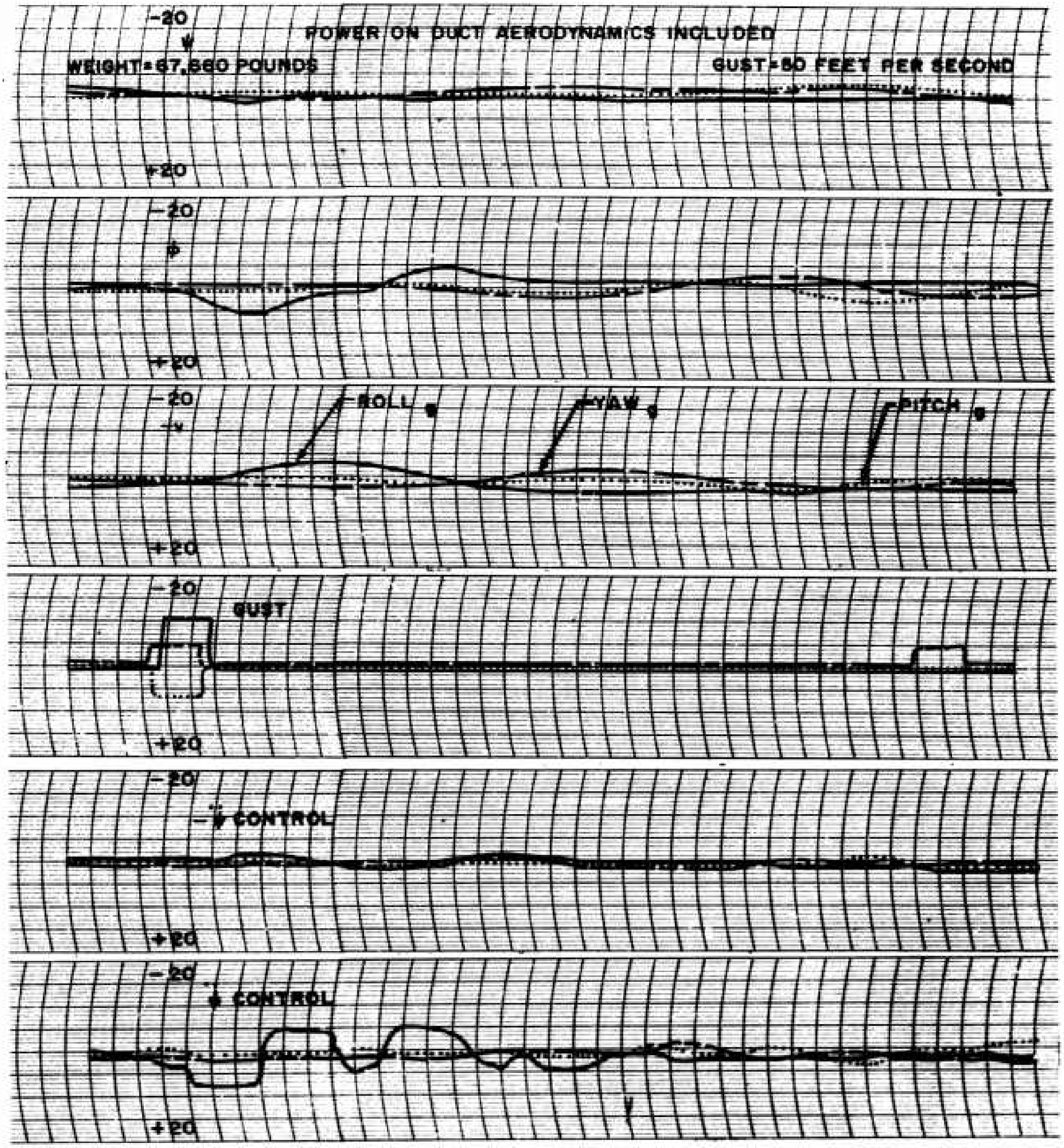


Figure 51a. Effect of Roll, Pitch, and Yaw Gusts in Hovering. Lateral Dynamics

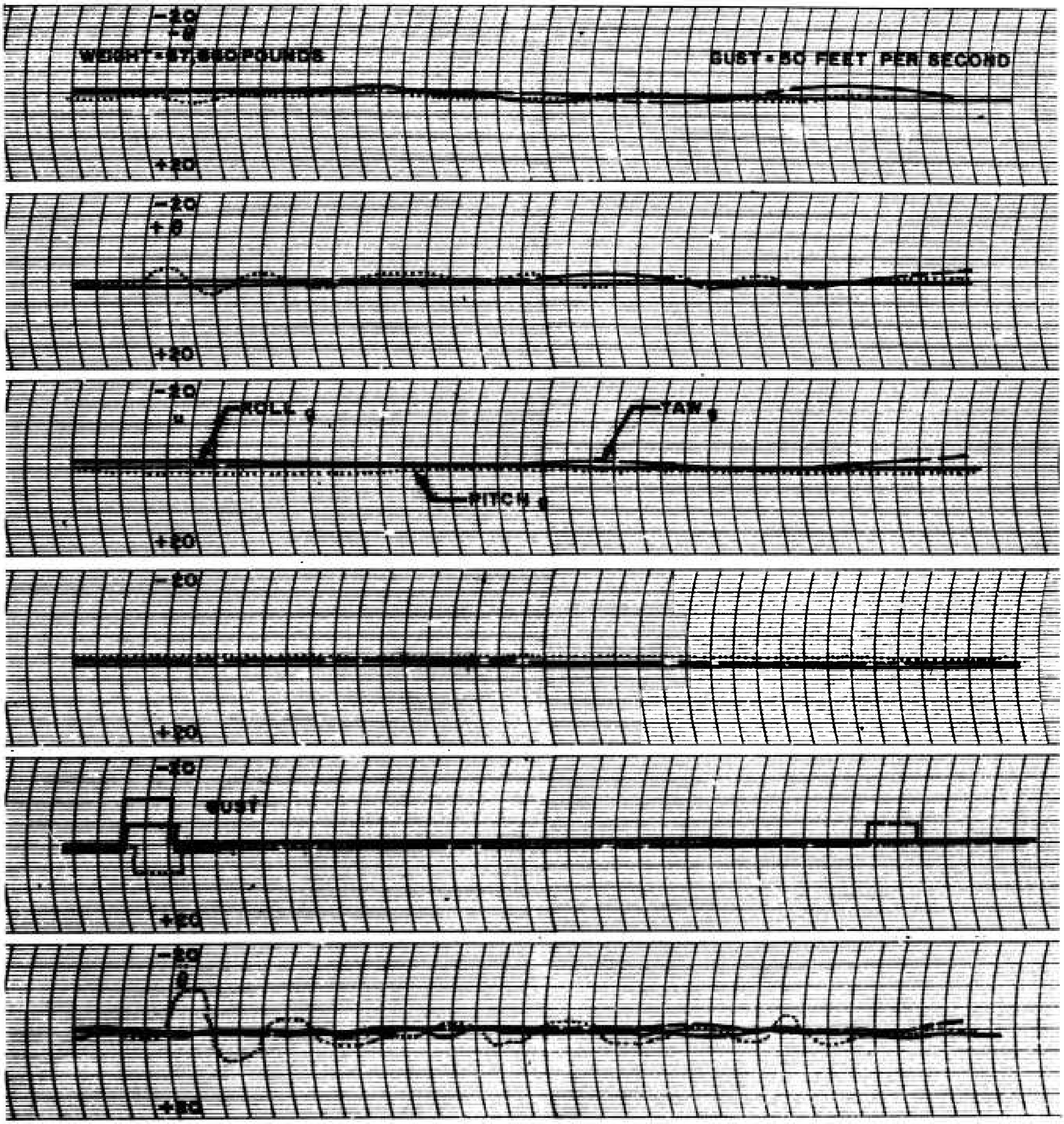


Figure 51b. Effect of Roll, Pitch, and Yaw Gusts in Hovering. Longitudinal Dynamics

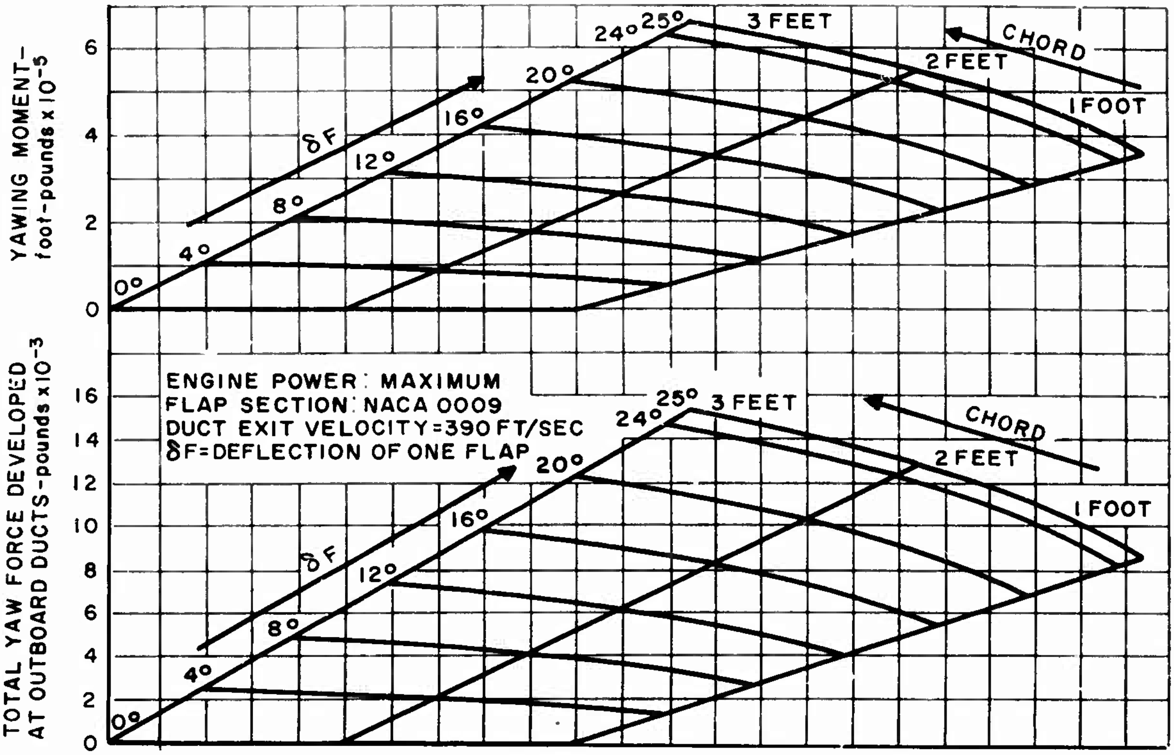


Figure 52. Yaw Reaction Control Forces and Moments Developed by Flaps at Exit of Outboard Ducts

5. The yaw controls for hovering flight can be obtained either from a ducted flap arrangement or a reaction engine at the tail.

6. The static longitudinal stability of the airplanes is ample and of proper order for service level flight conditions.

7. The lateral - directional stability of the airplane is ample throughout the level flight regime.

8. The inherent engine gyroscopic moments, tending to couple the pitch and roll planes of the airplane, prove to be quite controllable to the pilot for the hovering and transition cases studied.

9. The slipstream effects of the fan on the horizontal tail are negligible, due to the outboard location of the ducts.

10. The large moments of inertia of the airplane result in a relatively easily controlled stabilized airplane.

The following recommendations are suggested for future stability and control studies:

1. Detailed studies and a wind tunnel investigation of the duct flap controls made, for proper design refinement.

2. REAC studies of control lag made, for the dynamic hovering and transition cases, to determine the influence of such lag.

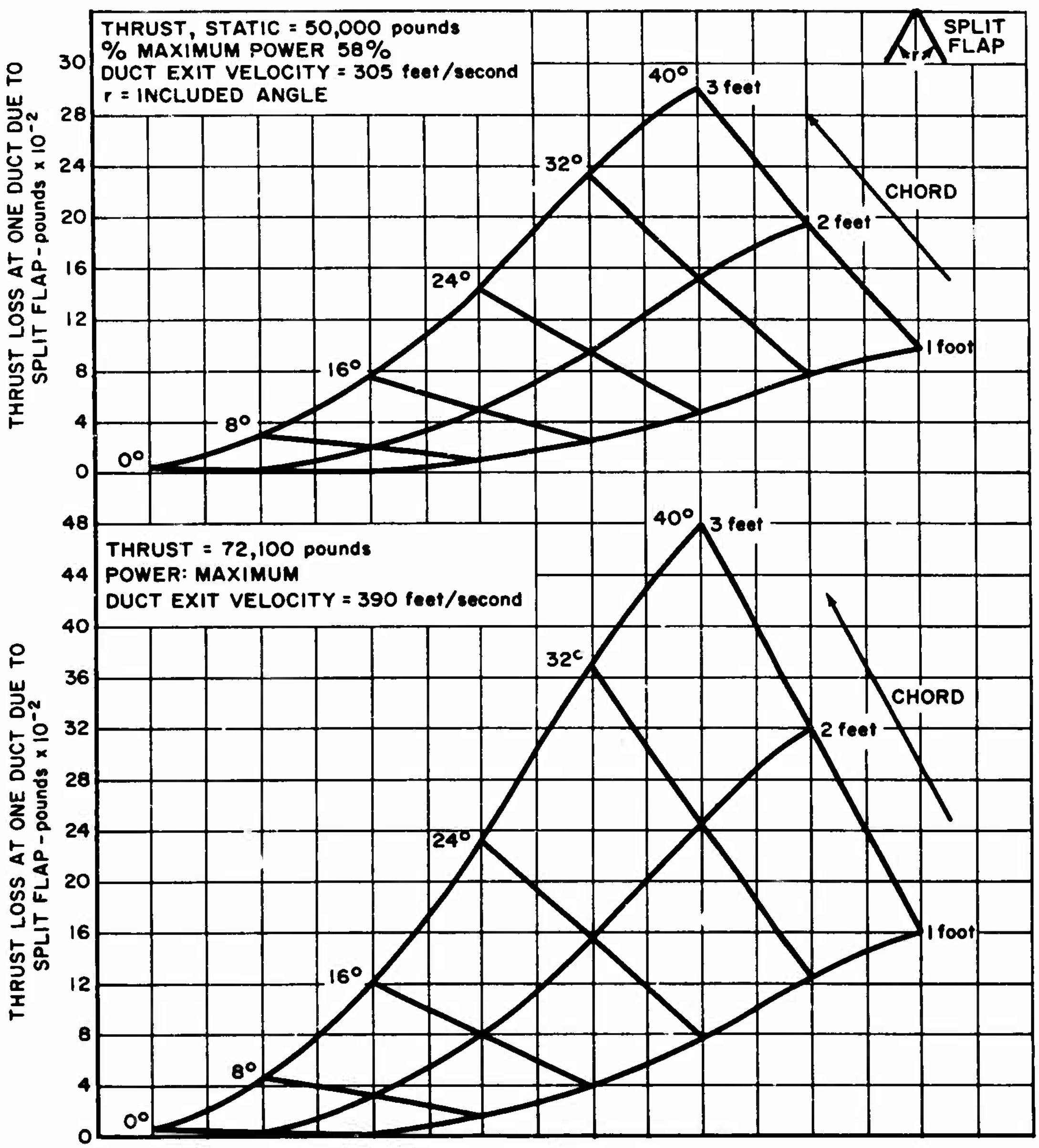


Figure 53. Roll Reaction Control Forces Developed by Split Flap at Exit of One Outboard Duct

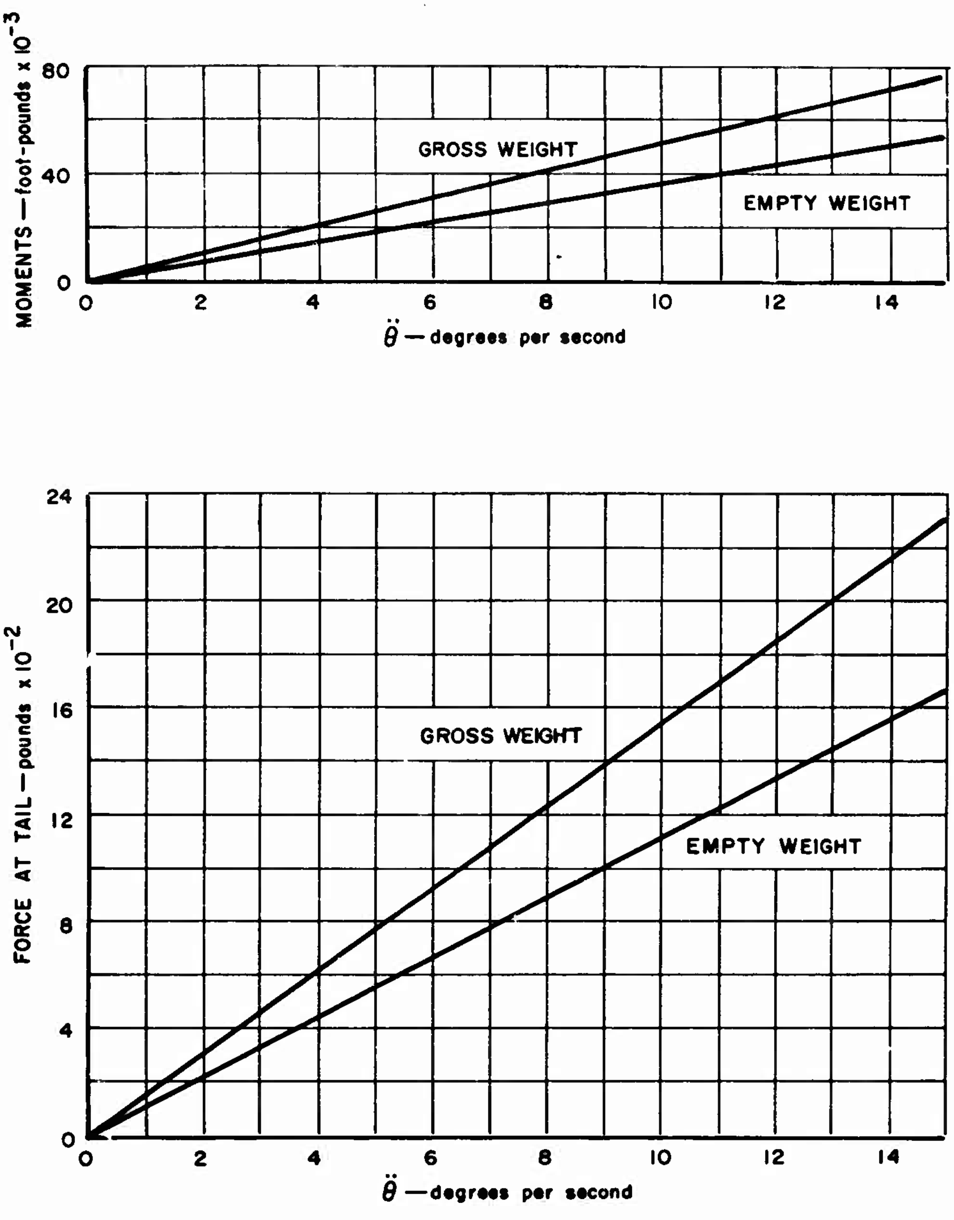


Figure 54. Pitch Reaction Control Forces and Moments Required for Various Pitch Gradients

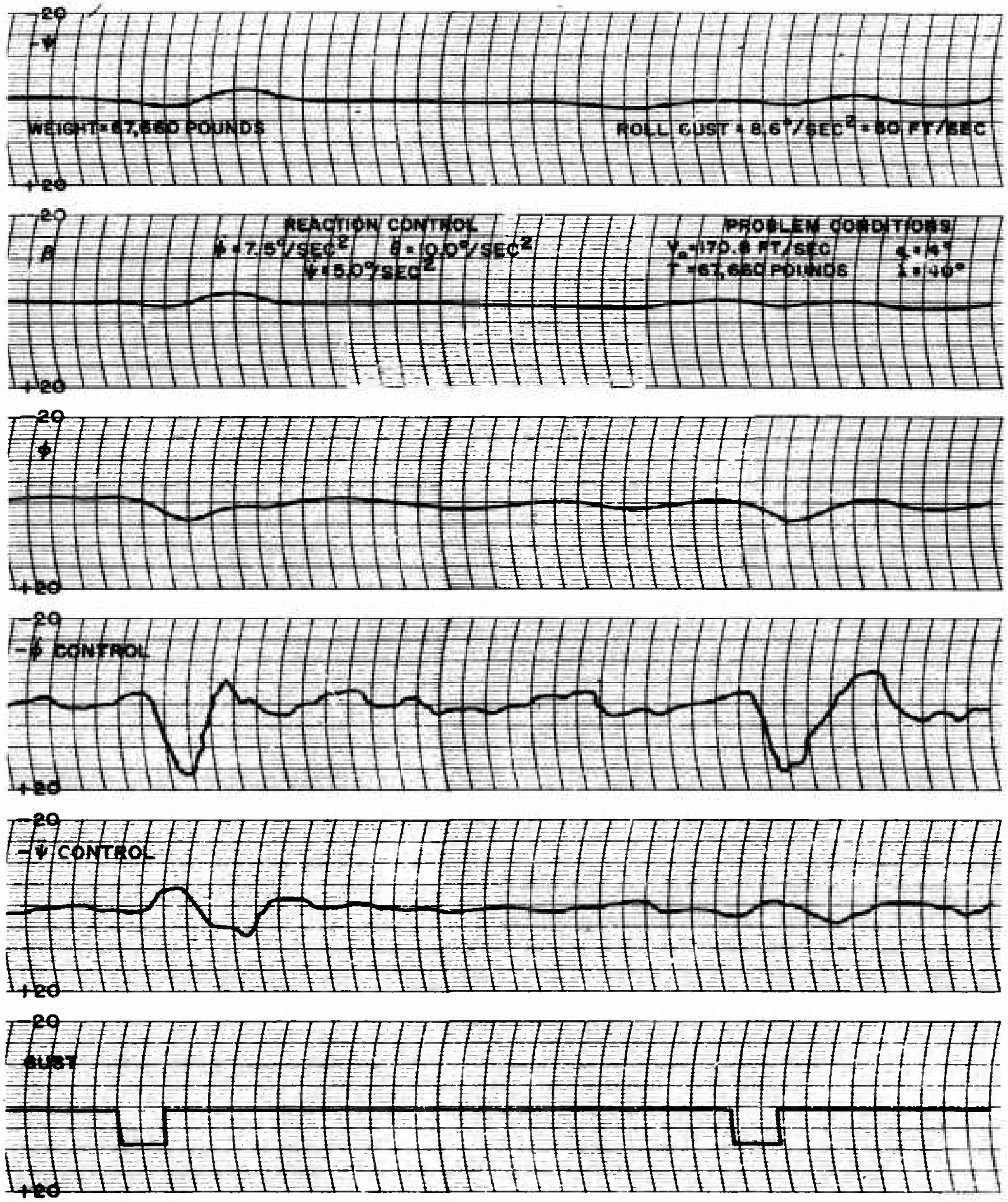


Figure 55a. Transition Lateral Traces

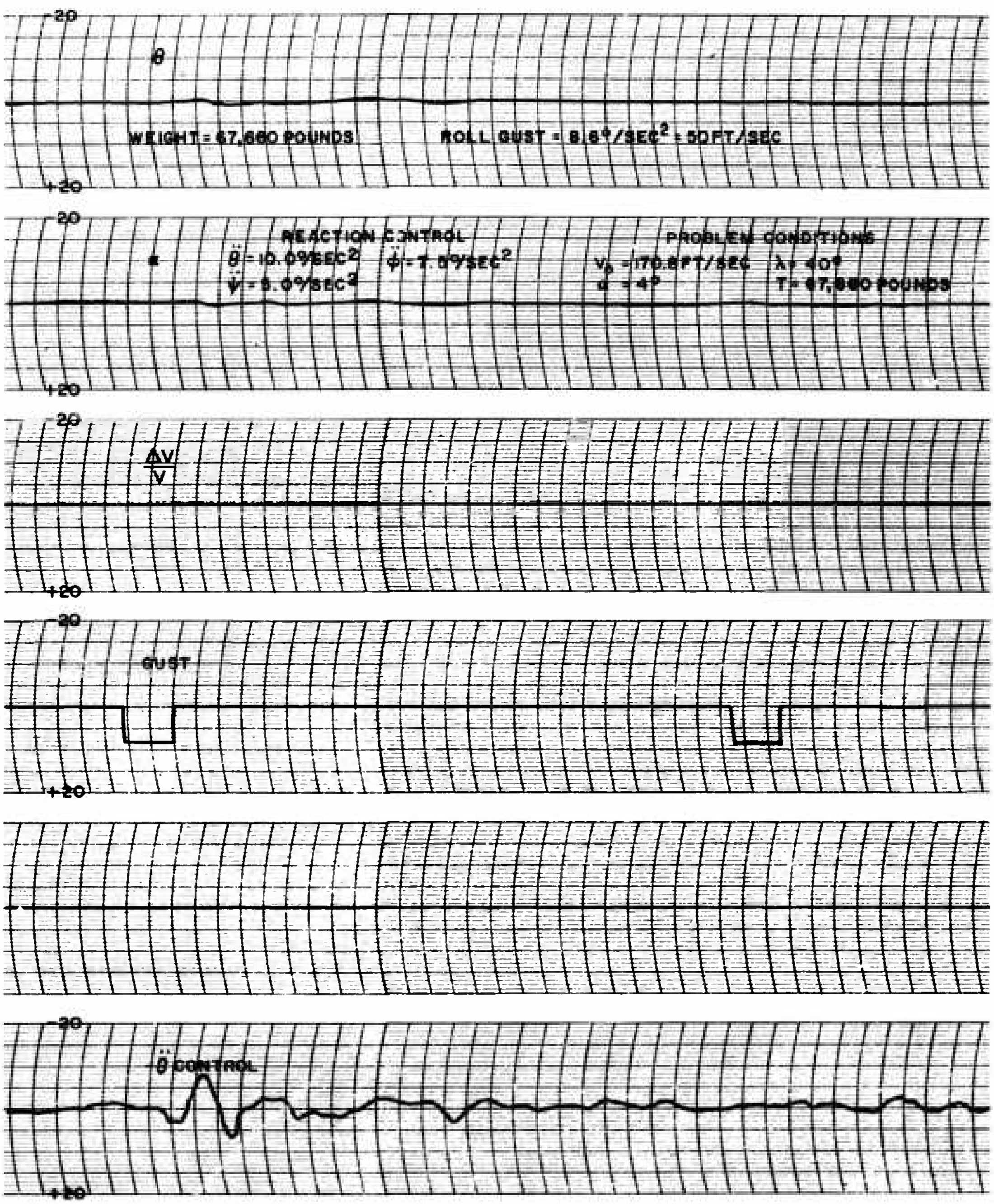


Figure 55b. Transition Longitudinal Traces

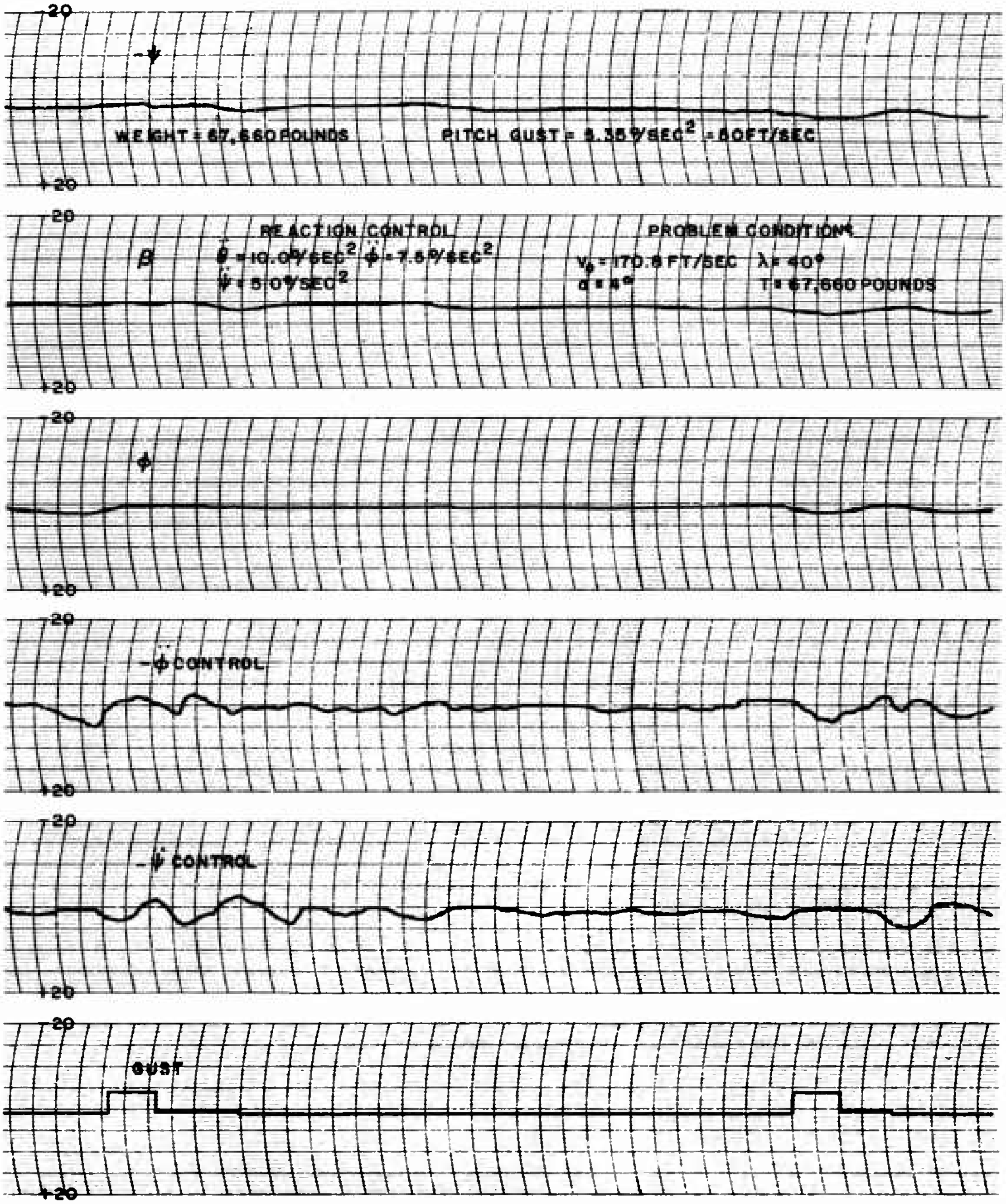


Figure 56a. Transition Lateral Traces

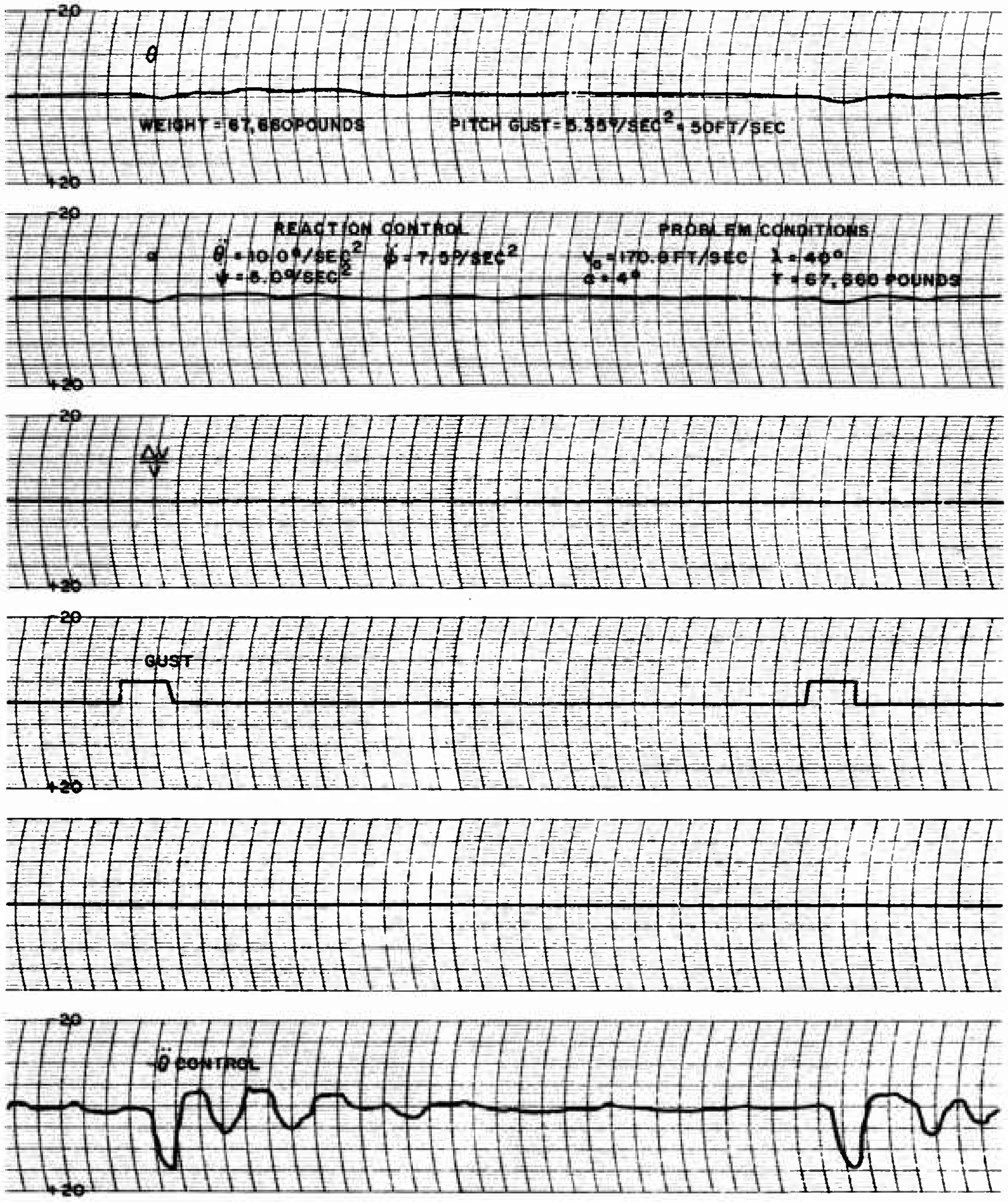


Figure 56b. Transition Longitudinal Traces

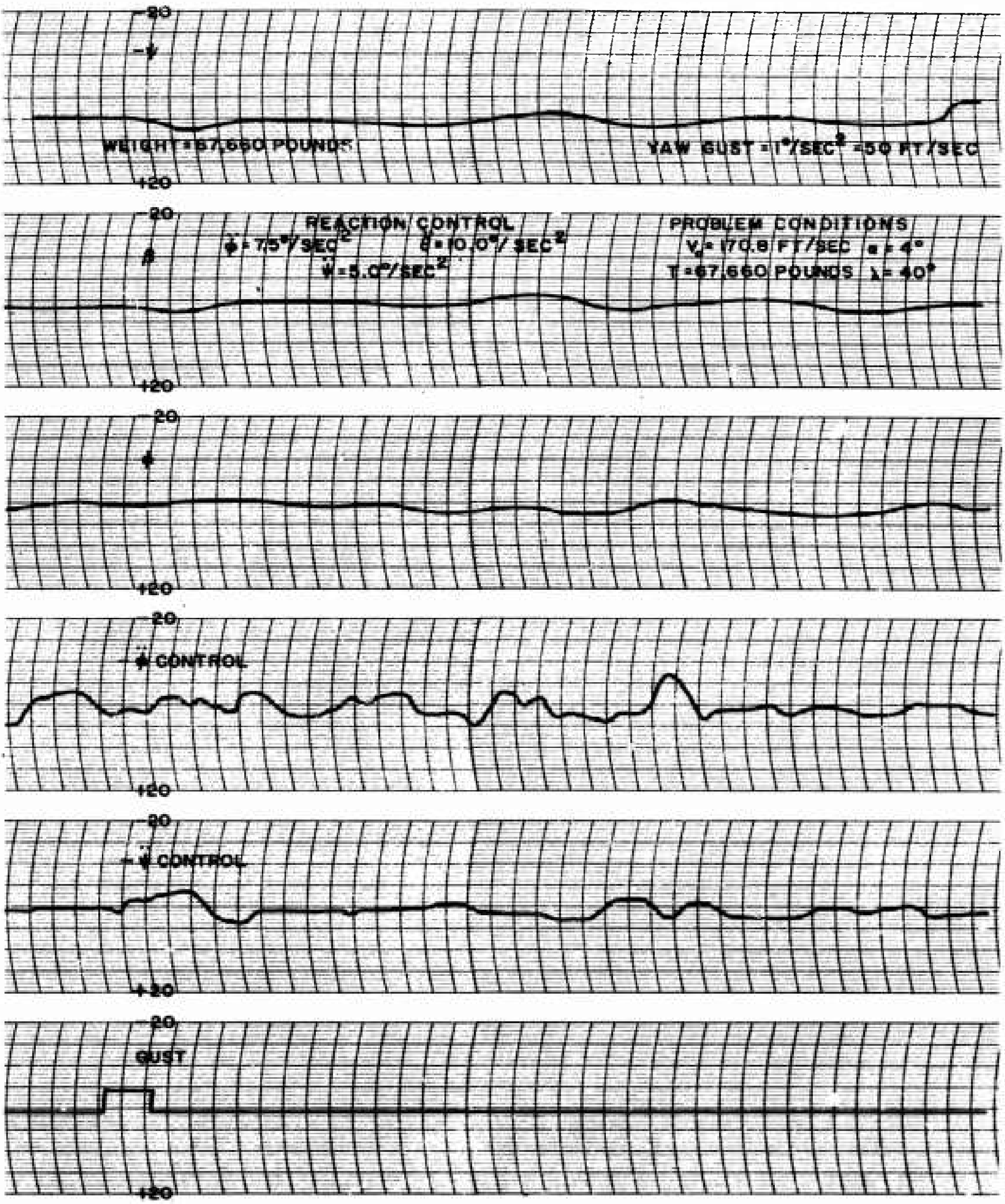


Figure 57a. Transition Lateral Traces

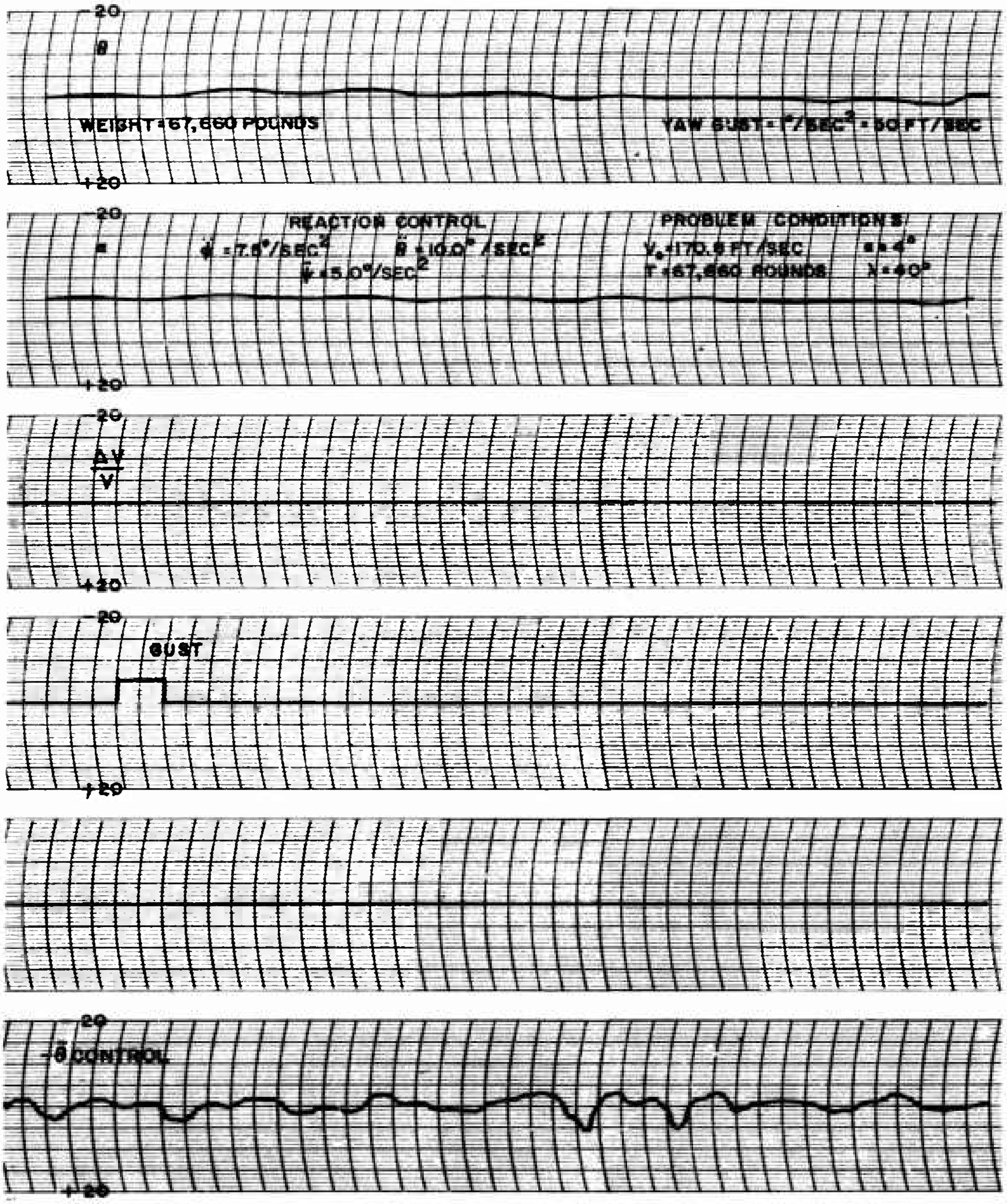


Figure 57b. Transition Longitudinal Traces

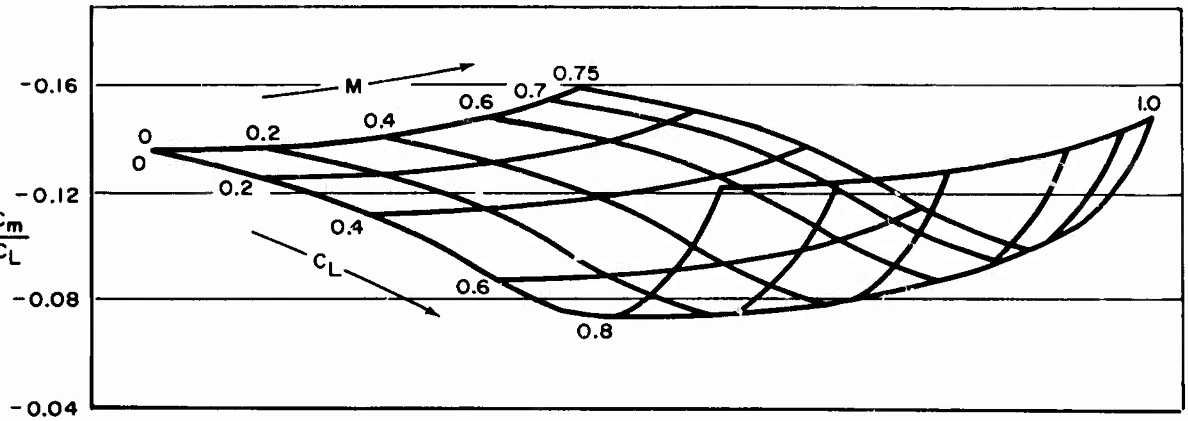


Figure 58. Airplane Longitudinal Stability

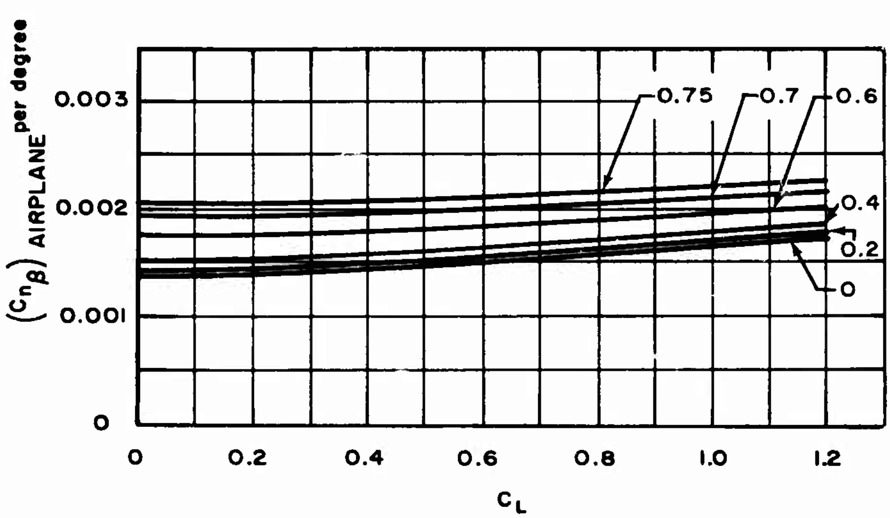


Figure 59. Airplane Directional Stability

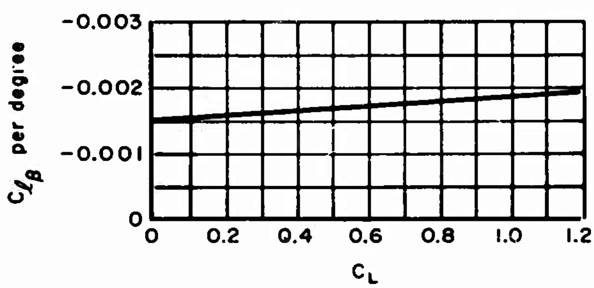


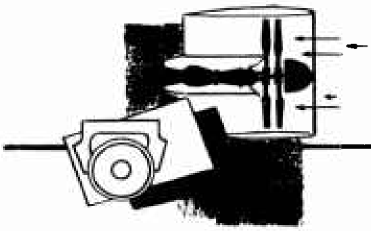
Figure 60. Rolling Stability of Airplane

3. A flight simulator program initiated, to include a complete flight from hovering through transition with pilot flying a given flight path.

4. A complete wind tunnel program initiated, to evaluate the interference efforts resulting from the ducted propeller on the aerodynamic characteristics of the airplane.

5. The study of power on and power off duct aerodynamics expanded, and given wind tunnel substantiation. (The first step in this direction is the wind tunnel test recently started by the University of Wichita.)

6. The influence of boundary layer control on duct inlet and exit sections should be evaluated.



Duct and Propeller Analysis

Bell Aircraft Corporation has conducted a systematic analysis of the effects of various design parameters on ducted propeller performance. A preliminary design procedure has been established, and the detail design of several different ducted propeller configurations has been completed. Wind tunnel tests now underway at the University of Wichita will provide experimental data which will be compared to the theoretical results of this study.

Momentum Study

A generalized compressible momentum study has been completed which shows the effects of duct geometry, horsepower input, and altitude on ducted propeller performance. Preliminary analyses such as the thrust variation with velocity shown in Figure 61 indicated that high subsonic speeds may be attained with this type of aircraft. This figure shows that the low pressure ratio fans suffer the largest decrease in thrust ratio as velocity increases. Detailed analyses have shown that, as demonstrated in Figure 62, for a given power input, there is an optimum static pressure ratio at which the best values of thrust/horsepower ratio may be obtained. The parameter HP/A_{in} is critical in determining the static thrust obtainable, lower values of HP/A_{in} resulting in higher values of static thrust/horsepower. Fan and exit area of the duct have been assumed identical in these studies to correspond with the assumption of cylindrical afterbody and lack of diffusion. The ratio of exit area to inlet area and the velocity at the fan are shown as variables of horsepower to inlet area ratio in Figures 63 and 64. A carpet plot of the thrust data is shown in Figure 65. Under forward flight conditions, it has been shown that low disk power loadings are desirable. However, low power loadings result in a greater range of fan velocities over which the propeller must operate as shown by Figure 66, thereby increasing the off design problems. Static momentum data were obtained for sea level and 6000 foot altitudes with fan efficiencies varying from 0.6 to 1.0. Level flight momentum data was obtained at various altitudes for a range of fan efficiencies from 0.7 to 1.0. A complete summary of the momentum studies carried out under this contract is presented in Reference. Judicious use of these data should enable the design of a practical ducted propeller system.

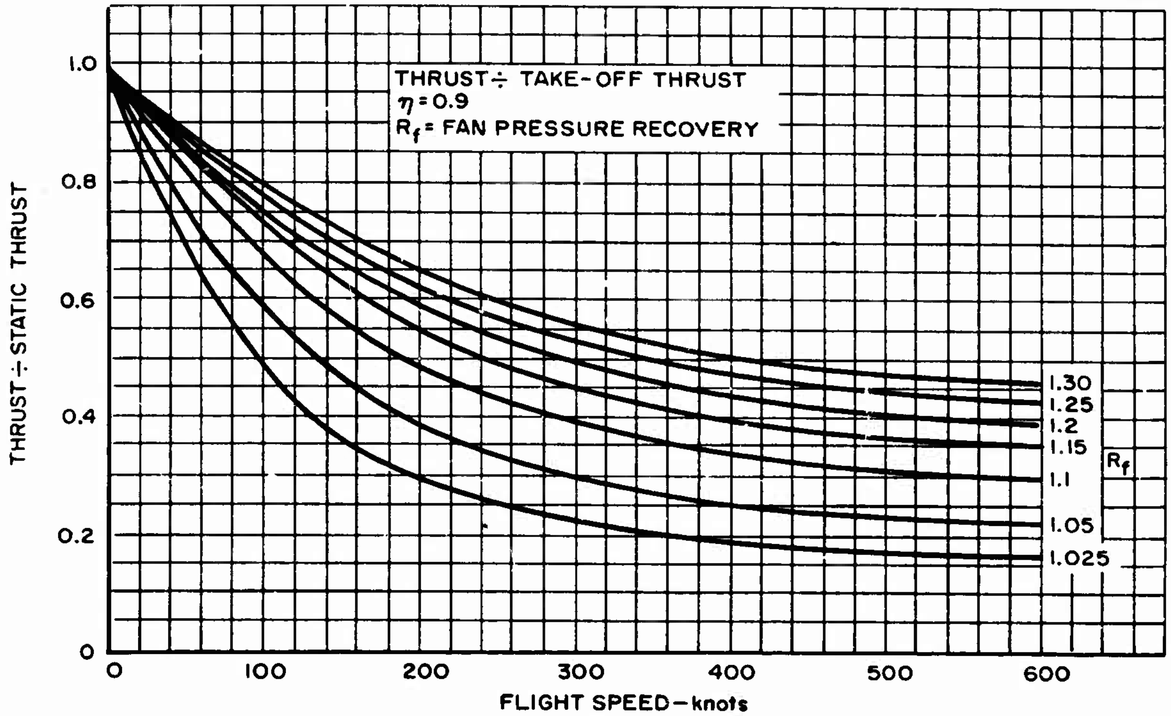


Figure 61. Typical Shaft – Power Turbine plus Fan Characteristics

Turboprop Engine Survey

In the design of a ducted propeller system, it is essential that the duct design and engine selection be carried out simultaneously, since the internal geometry of the duct and the power available are critical factors in determining the amount of thrust available from the system. All of the ducted propeller configurations considered during this study have been designed in conjunction with a specific turboprop engine. A turboprop engine is the most logical powerplant for ducted propeller applications since it has a low weight/power ratio available and a relatively small frontal area. A survey of the foreign and domestic powerplants available was completed and is presented in Table VIII. For the particular engines used, the variation of the horsepower thrust and fuel flow characteristics with speed and altitude are presented in Reference 2.

Ducted Propeller Studies

Duct design has been carefully considered during the study. In order to resolve the conflicting requirements which necessitate a large bellmouth inlet under static conditions and a low drag configuration for high speec flight, retractable inlet flaps have been utilized in all designs. The arrangement and actuation of these flaps have been discussed in the preceding section on design studies. It is expected that good results will be obtained from the resultant split-flap arrangement, and this assumption is supported by the experimental results of Kruger in Reference 15. The effects of shroud length and hub-tip ratio were also investigated. In the reference cited, Kruger has shown the desirability of small shroud length/diameter ratios. A large hub-tip ratio has the advantages of less blade twist, less sensitivity of the system to off-design operation, and large hub volume to permit

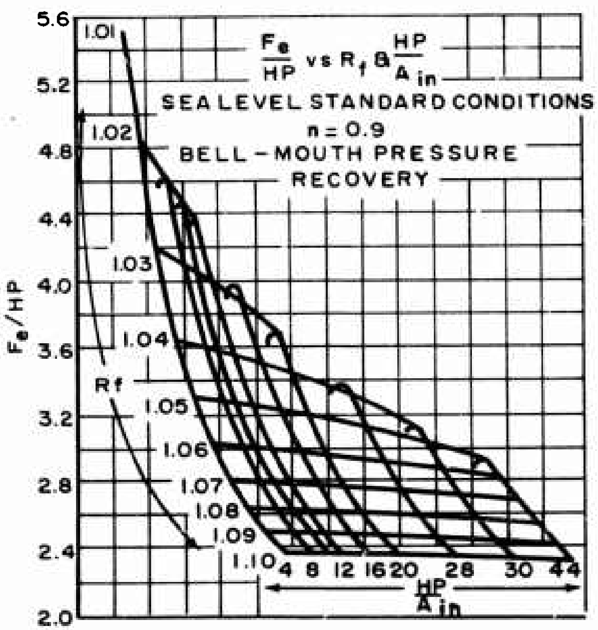


Figure 62. Static Momentum Data

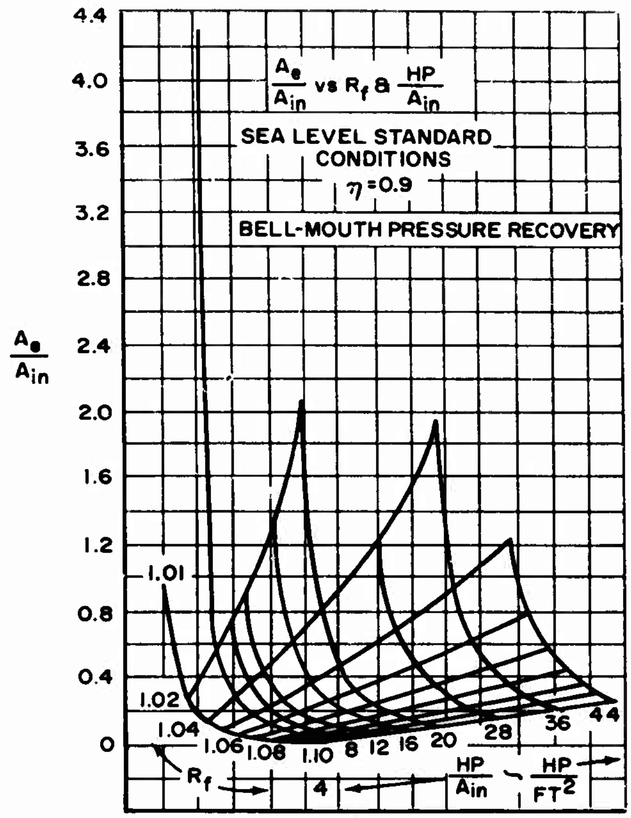


Figure 63. Static Momentum Data

engine installation in the hub or the use of multi-bladed variable pitch mechanism. Disadvantages of the large hub, of course, are the weight and the adverse effects of drag. These factors must be compromised in the selection of hub size.

Due to the over-all size limitations imposed upon the ducted propeller by weight, performance and stability, it was judged advisable to employ high fan power loadings. These high power loadings resulted in propeller solidities and pressure ratios which were much higher than those used in conventional practice. As a consequence, low speed cascade data, such as that used in the design of axial flow fans and compressors, was the most reasonable source of data for the detailed blade design of the ducted propeller configurations presented in this report. Three major propeller configurations were studied. The first configuration used a fixed, high solidity rotor together with variable inlet guide vanes and fixed exit stators. The second design used a variable pitch propeller in conjunction with fixed exit stators. The third configuration consisted of two contrarotating variable pitch propellers. A modified NACA 65 series compressor blade section has been used for rotor design. In those instances where solidities lower than those covered by the available NACA data have been used, the available data has been extrapolated to cover the lower solidity range. These extrapolations have been checked by brief experimental tests conducted by the NACA and found to be in fairly good agreement at the design point. At off-design points, however, agreement was poor. It is felt that the data thus obtained is valid as used in this study, since the low solidity data was used with the variable pitch configurations

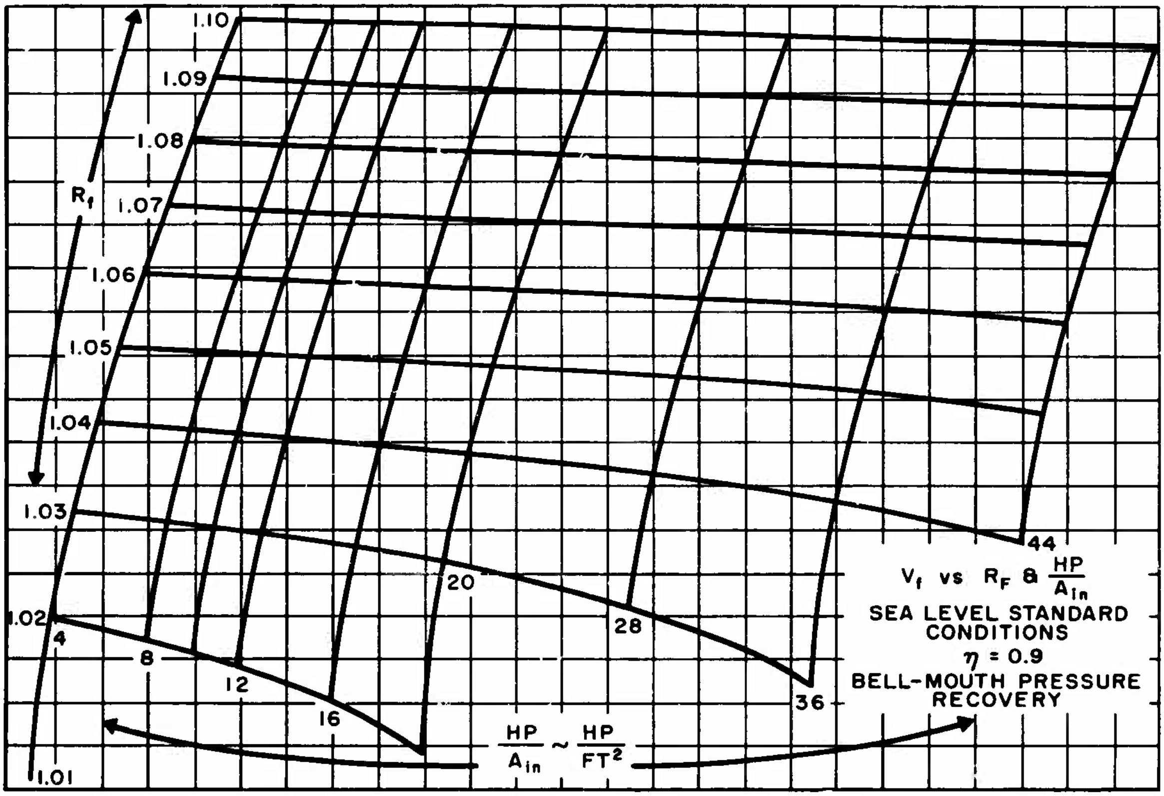


Figure 64. Static Momentum Data

where the blade angle of attack variation over the speed range was not as great as for the fixed pitch configuration. Further cascade tests in the low solidity range are desirable. The C-Factor of Reference 16 was used to check blade limit loading, and the data of this Reference has been replotted in carpet form. A linear variation in blade thickness has been assumed, the variation being from 10 percent at a relative inlet Mach number of 0.4 to 4 percent at a relative inlet Mach number of 1.0. In the configurations which employed exit stators, the exit stators have been designed to remove all of the residual whirl in the airstream under static conditions at 6000 feet on a 95° day.

The design of a ducted propeller system which will perform satisfactorily at a given point in the flight regime is relatively straightforward, but operation at other than the design point presents several problems. At other points in the flight regime the forward velocity, altitude, horsepower input, rotational velocity, or a combination of these parameters will be different than at the design point. Under these conditions, a configuration will operate inefficiently if the blade pitch is fixed, and over a large range of operating conditions, will overspeed or suffer a drastic reduction in thrust. With a direct-drive constant speed engine such as turboprop, these conditions cannot be alleviated by varying the engine rpm. The most practical means of solving the off-design problem is to incorporate some means of varying the effective angle of attack of the blades, such as the use of the variable pitch or variable inlet

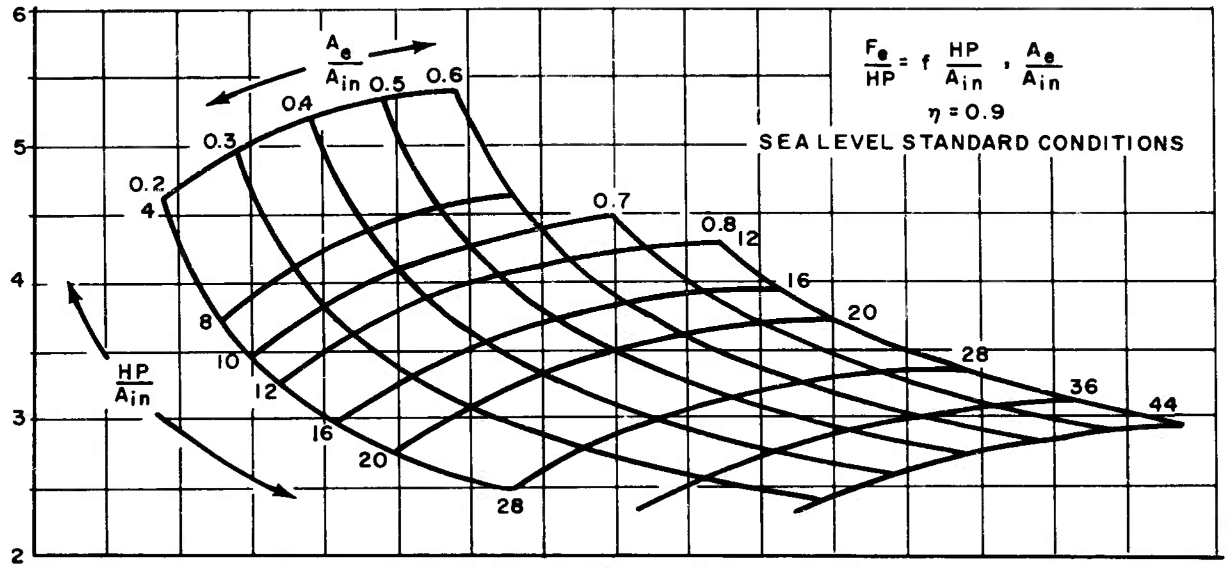


Figure 65. Ducted Fan Performance – Static

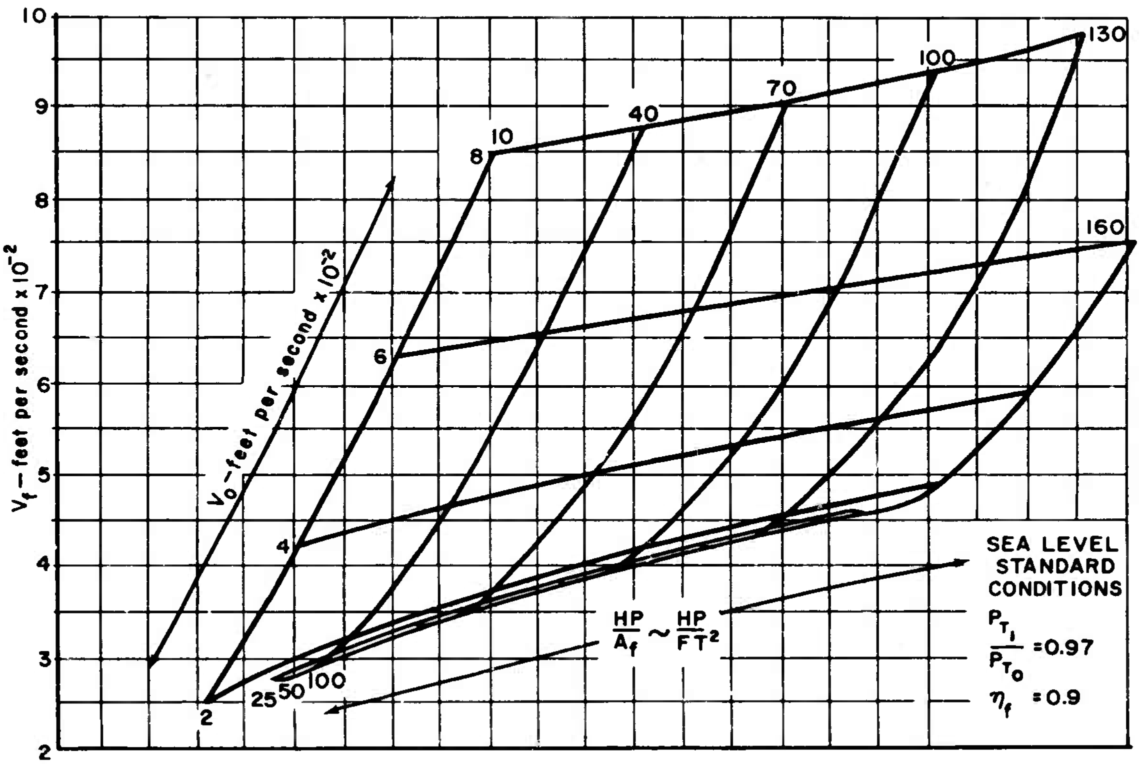


Figure 66. Inflight Ducted Propeller Performance. Axial Velocity

TABLE VIII

SUMMARY OF TURBOPROP ENGINES

Manufacturer	Designation	T. O. Shaft HP	Weight lb	Weight Power lb/HP	Thrust lb	Thrust Weight	SFC lb/Ht/S.HP	Length in.	rpm T.O.	Power Turbine	rpm at Partial Power	Reference	Date of Issue
American Engines													
Allison	[1] T54 (500 - C14)	7510	3150	0.420	1480	0.470	0.560	123.6	13820	Coupled	13820 at 60%	T54 Brochure	May 1955
												Spec. 382	Apr 1954
	T56 (501 - D8)	3755	1785	0.475	740	0.415	0.570	145.0	13820	Coupled	13820 at 50%	Spec. 377C	Apr 1955
	550 - B1	5168	2150	0.416	830	0.386	0.543	137.0	9900	Two Spool	8670 at 60%	Spec. 394B	Jan 5, 1956
Boeing	T50	210 (ESHP)	240	1.140			1.3 (per ESHP)	40.0				Aviation Week	May 12, 1956
	502 - 10B	270 (ESHP)	245	0.910			1.02 (per ESHP)	43.0				Aviation Week	May 12, 1956
Continental	XT51-T-3 (Artouste II)	425	236	0.555	75	0.318	0.970	44.3	6000	Coupled	6000 at 60%	Spec. 2110	Sept 3, 1954
	[2] Turmo II (260 - 2)	396	384	0.970	60	0.156	1.050	57.2	6000	Free	5020 at 60%	Spec. 2087	Dec 1953
General Electric	T58-GE-2	998	325	0.326			0.660	59.0	19500	Free		Spec. SE-1	Sept 9, 1954
Lycoming	T53 (LTC 1B)	825	460	0.560	102	0.222	0.710	48.0		Free		Spec. 104.2	Oct 15, 1955
	T53 (LTC 1C)	960	460	0.480	118	0.256	0.680	52.0	26100	Free		Spec. 104.1	Aug 3, 1955
	T55 (LTC 4B-1)	1600	600	0.375	190	0.317	0.680	45.0	18100	Free	16630 at 75%	Spec. 124.4	Jan 16, 1956
Pratt & Whitney	T34 (T34-P-3)	5500	2590	0.471	1250	0.483	0.687	157.0	11000	Coupled	10250 at 50%	P & W Inst. Hdbk. (T34)	Apr 28, 1954
	T57 (YT57-P-1)	14000	6600	0.471	5000	0.758	0.570	196.0	6525	Two Spool	5300 at 50%	P & W Inst. Man. Prel. Perf. Data	Aug 15, 1955
Westinghouse	[3] R. Da. 7/2	1910	1201	0.630	500	0.416	0.730	98.0	14500	Coupled		Westinghouse R & T A1840	
												R. R. Sheets Sept 21, 1955 &	Oct 4, 1955
	RB-109	4020	1875	0.466	1000	0.533	0.505	98.0	14100	Two Spool	11000 at 50%	R. R. - T. S. D. Pub. 499	June 1953
	[4] RB-109	4020	1935	0.481	1175	0.610	0.567	100.3	14100	Two Spool	11000 at 50%	Brochure	Feb 10, 1956
Wright Aeronautical	T47	11400	4825	0.423	3050	0.632	0.570	160.0	7700	Two Spool	6800 at 50%	Wright SP 876	July 20, 1951
	T49 (YT 49-W-1)	9000	4466	0.496	3450	0.772	0.758	102.0	8000	Coupled	7350 at 50%	Wright Spec. 875E	Jan 4, 1954
Foreign Engines													
Armstrong-Siddeley	Mamba	1270 (ESHP)			375		0.920						
	Double Mamba (ASMD.3)	3035 (ESHP)	2200				0.75 (per ESHP)	101.0				Aviation Week	Mar 12, 1956
	Python	4100											
Bristol	BE-25	4000	3020	0.755	2730	0.905	0.815	110.0		Two Spool		BE-25 Ser. 1 Eng. Spec.	
Napier	Eland (N.E.L. 4)	3785	1800	0.478	610	0.339	0.595	121.0	12500	Coupled	10900 at 50%	Brochure P.S. XL118	Nov 1953
Rolls-Royce	Dart	1640	1157	0.705	365	0.315	0.750	96.0		Coupled		R.R. - T.S.D. Pub. 500	
	RB-109	4020	1835	0.456	1000	0.545	0.505	98.0		Two Spool		R.R. - T.S.D. Pub 499	June 1953

1. Rating to increase to 9000 ESHP by 1961 (T54 Brochure dated May 1955)
2. Turmo III prototype Max. rating 750 SHP at 0.78 SFC (Engine Summary dated Sept. 25 1955)
3. Advanced R. Da. 8 will have a Max. ESHP rating of 2500 HP.
4. RB-109 has growth potential to 5000 SHP by 1964.

guide vanes which have been examined in this report. A simplified expression which indicates power absorption at off-design conditions, called the power factor, has been developed. This factor indicates the inlet guide-vane turning or pitch change necessary to absorb the full power input.

Specific Design Studies

The designs are best considered in conjunction with the particular powerplants they utilized. Three specific engines were used in the ducted propeller systems designed under this study. They are the Wright Aeronautical T49, the Rolls Royce RB-109 and the Allison 550-B1. The T49 configurations were fixed pitch-rotor configurations, employing variable inlet guide vanes and exit stators. This configuration was also used with the RB-109 engine. In addition, the single rot tion variable pitch configuration employing exit stators was also designed for the RB-109 engine. The third method of varying the geometry was evaluated by examining a contrarotating propeller designed for use with the Allison 550-B1.

The most satisfactory T49 configuration has a design flight speed of 200 knots at sea level standard conditions. The fan diameter is 11.6 feet and the inlet diameter under static conditions is 15.5 feet. This design has 17 fixed-pitch rotor blades, 19 exit stators, and 21 symmetrical variable-pitch inlet guide vanes. The power factor, an indicator of power absorption at various off design positions, has been plotted versus guide vane setting for a range of conditions in Figure 67. The fan rotates at 1235 rpm, the power available at the design point is 9383 HP, and will put out a static thrust of 17,370 pounds at 6000 feet on a 95° day. The inlet guide vane settings range from +18° to -15° at sea level, as is shown in Figure 68. While this fan is not an optimum design, it is

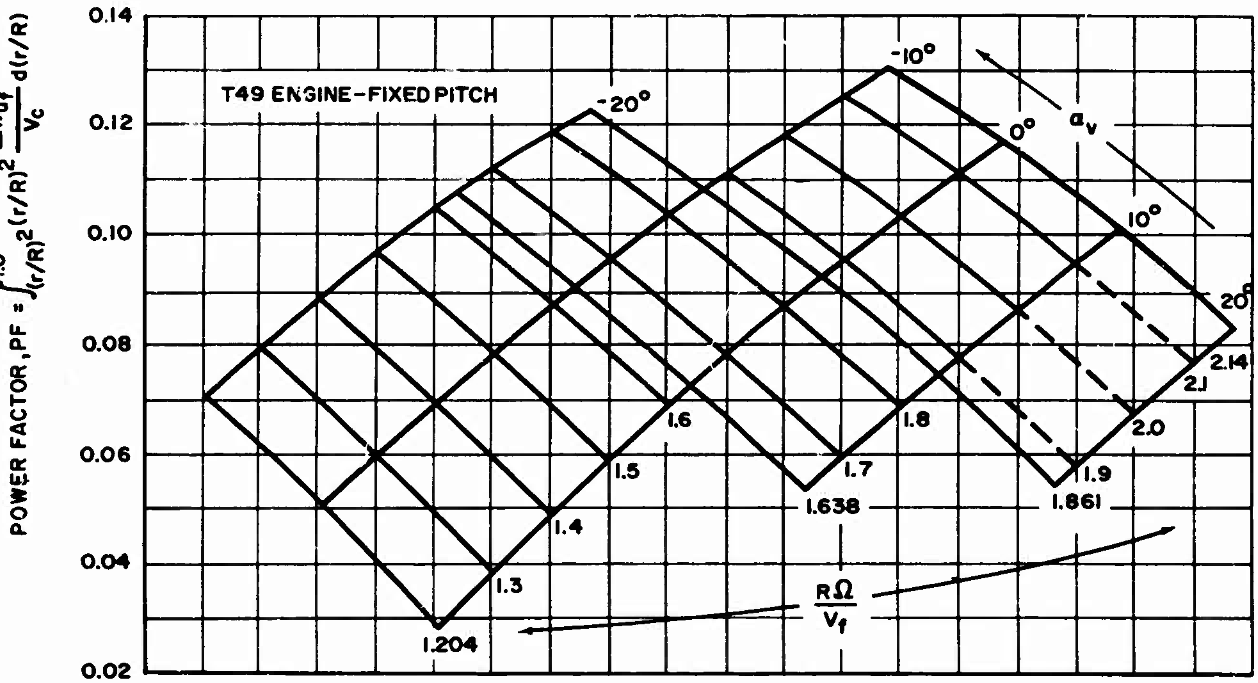


Figure 67. Power Factor Configuration No. 1

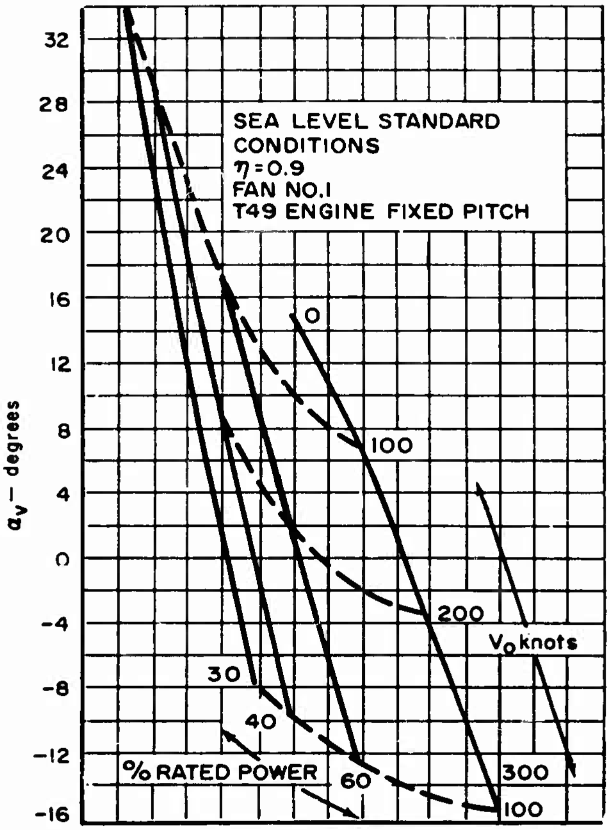


Figure 68. Inlet Guide Vane Angle Required at Various Settings

an acceptable configuration which will operate satisfactorily over the required speed range.

The best RB-109 design was a variable pitch configuration having a design point of 200 knots at sea level. This configuration has 10 variable-pitch rotor blades and 15 fixed exit stators. The power factor variation with pitch settings for the range of operating conditions is shown in Figure 69. The fan and inlet diameters are the same as for the T49 design, but the hub is somewhat smaller, the hub-tip ratio being 0.3 as compared to 0.4 for the T49. The fan rotates at 1143 rpm and is designed for a static thrust of 11,317 pounds at 6000 feet on a 95° day. The required pitch change ranges from approximately 10° under static conditions to -16° at a forward speed of 400 knots. The range of pitch settings required for level flight are shown in Figure 70.

The Allison 550B-1 engine was evaluated in conjunction with a fixed pitch, inlet guide vane configuration and a pair of contrarotating fans. Several designs having inlet vanes using varying degrees of pre-whirl at the design point were also analyzed, but were not detailed due to a lack of contract time. The contrarotating propeller configuration was analyzed in detail to evaluate the merits of this type of arrangement. The propellers had a hub-tip ratio of 0.5, thereby permitting the installation of two Allison 550-B1 engines in the hub. The pitch variation with power factor for the front rotor is shown in Figure 71 and for the rear rotor in Figure 72. The fan diameter of this design is 11.8 feet and the static inlet diameter is 15 feet. The contrarotating propellers turn at 996 rpm and are designed for a total static thrust of 22,550 pounds at the high altitude and temperature conditions. Assuming that the maximum forward flight speed is 460 mph at sea level, the pitch change required for the front rotor is 10.25° and for the rear rotor it is 11.75° as is shown in Figure 73.

Duct and Propeller Study Results

As a result of the studies contained in this report, it appears that both inlet guide vanes and variable pitch are feasible means of obtaining satisfactory off design operation. A variable pitch configuration is somewhat more desirable, since the range of the variation in blade effective angle of attack is smaller. Although the incorporation of variable pitch into a fan having 10 or 12 blades might necessitate an extensive development program, the large volume of the hub should be adequate to house a practical variable pitch mechanism. At the same time, since the ducting permits a smaller pitch variation as well as a smaller diameter in conjunction with a larger number of blades, to yield lower blade loadings, the variable pitch problem should be alleviated to some extent over conventional practice. With the inlet guide vane designs, which were based on

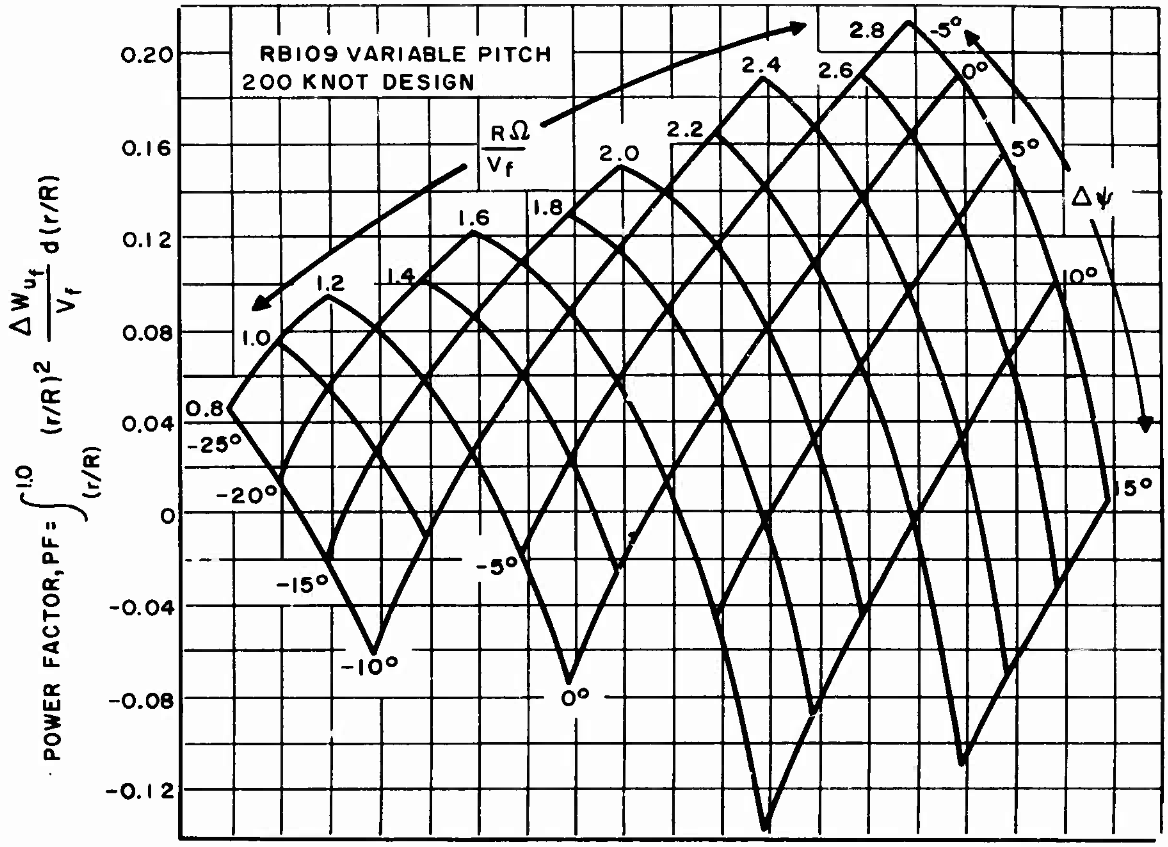


Figure 69. General Fan Performance Pitch Angle Variation

zero pre-whirl at the design condition, the high speed end of the flight regime has been somewhat limited due to the large vane angles required. It is felt that a further study of the pre-whirl concept will result in modifications which will enable these configurations to attain forward speeds fully competitive with the variable pitch designs.

The office of Naval Research has sponsored a wind tunnel program at the University of Wichita to provide some experimental verification for the studies completed in this report.

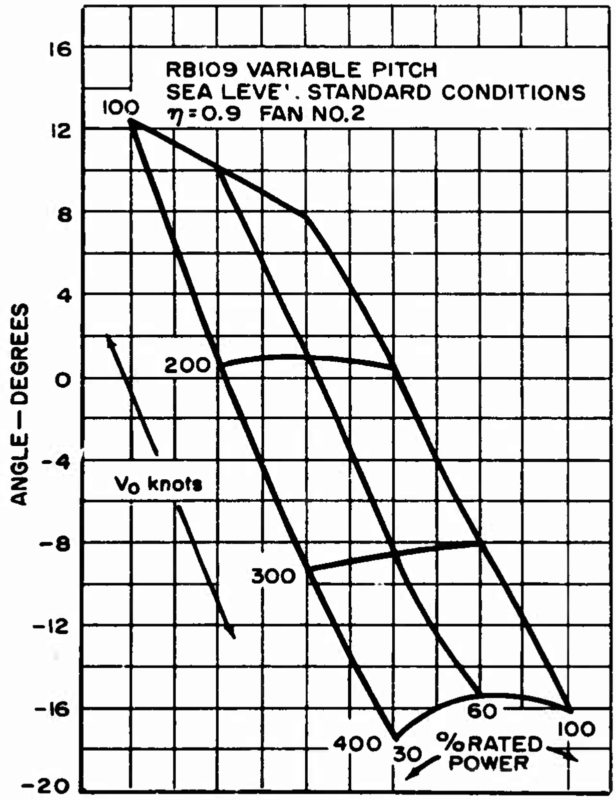


Figure 70. Pitch Change Required at Various Power Settings

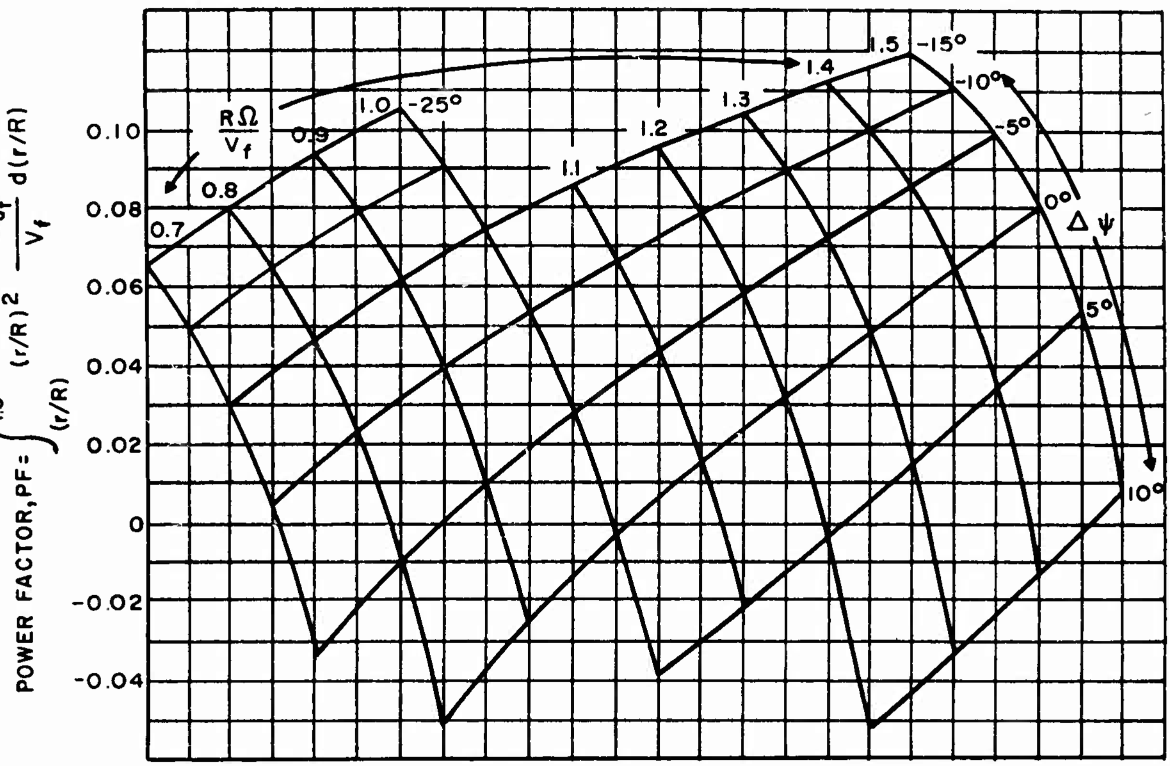


Figure 71. Front Rotor Performance for Allison 550-B1

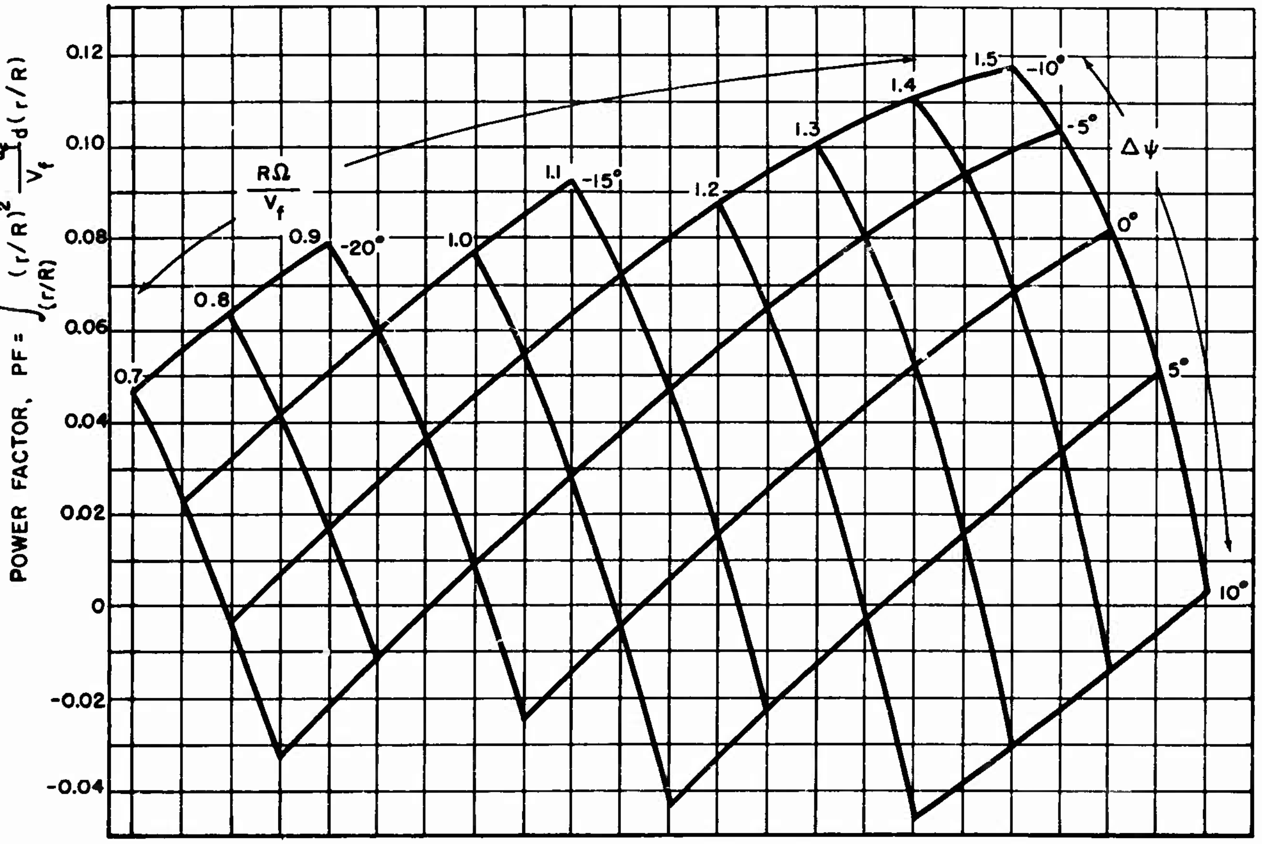


Figure 72. Rear Rotor Performance for Allison 550-B1

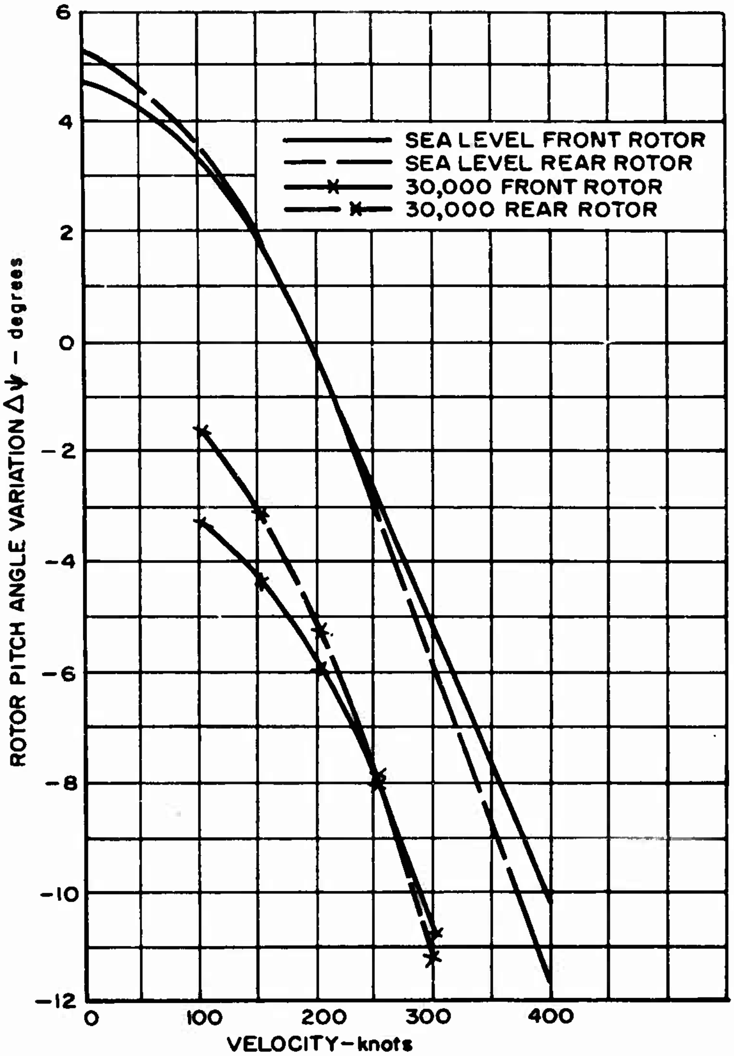


Figure 73. Contrarotating Propellers. Pitch Variation

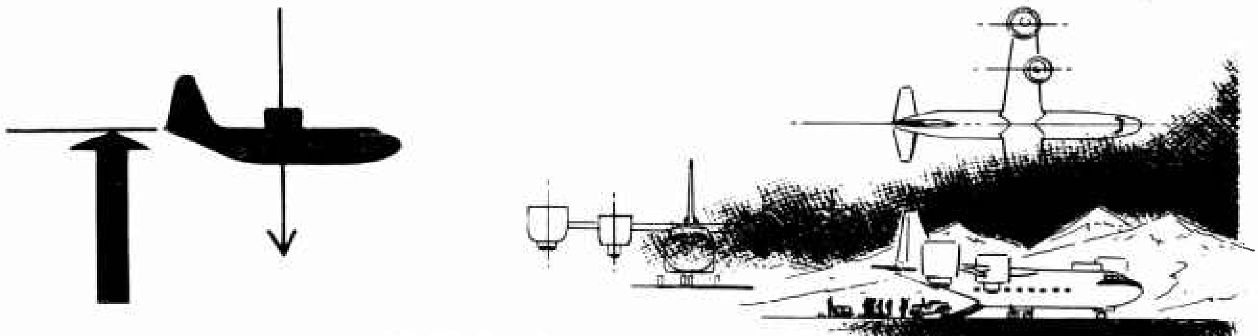
Survey Material

A review of the literature and conferences with interested agencies, bearing on the theory and design of ducted propellers were completed as part of this Contract. The survey has been reported in Reference 13. The literature review is divided into two main groups which treat the problem from the viewpoint of classical propeller analysis and as compressors. Since the distinction is not always clear cut there is necessarily some overlap in the division. The results of conferences with the various interested agencies is contained in a third section. In all, about 200 reports were reviewed of which about 75 were abstracted and included in this summary. Conferences were held with various agencies and companies to further extend the review and reports of these conferences were included.

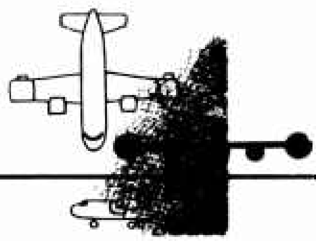
The review of literature indicated that while serious though was given to ducted propellers during World War II, the advent of jet propulsion cut most of the work short with the result that very little experimental data was available. Of the 75 reports abstracted only 17 dealt directly with ducted propellers and of these only 8 contained experimental data. For the most part these data were the first step in investigating a new idea. The tests covered a limited range of variables and were in general, sufficient to show the desirability of a complete program. The majority of the reports concern fans and compressors, and the methods are often applicable to ducted propeller work. The presentation of these abstracts is made under two major headings. The first section contains the reports which treat the problem from the viewpoint of incompressible flow and classical propeller blade element analysis. Included in this section are reports on ducts alone, ducted propellers, isolated airfoils, and lightly loaded fans. The second section of the bibliography contains the report which treats the problem from the viewpoint of compressor theory and practice. This group includes reports on compressors, on airfoils in cascade, and on the flow problems related to compressor design. Each of the major sections was subdivided into a section of theoretical reports and a section of reports containing experimental data. Within each subsection the reports are listed alphabetically by the author. A general breakdown of the literature reviewed might be made as follows. First, a group of 44 reports which considered the subject from the standpoint of incompressible flow and classical propeller analysis. Of these reports, 20 contained experimental data. The subjects covered and the number of reports in each category were: Wind tunnel fans 9, conventional propellers 3, ducts 13, ducted propellers 17, and isolated airfoils 3. A second grouping was made which dealt with compressors and treated the flow as compressible. There were 29 reports in this group of which 19 were experimental. The data of many of these reports was applicable to ducted propeller work. In general there exists a lack of systematic experimental data for use in design work. The theoretical coverage, while extensive, lacks the correlation of experiment and is therefore often unreliable for detail design.

The conferences on ducted propellers added substantially to the total information. Visits were made to the NACA Laboratories at Langley Field, Va. and Cleveland, Ohio, to the University of Wichita in Kansas, to the Navy Facility the David Taylor Model Basin, Washington, D.C., to Princeton University in New Jersey, to Kaman Aircraft Corp., Bloomfield, Conn., to Hamilton Standard Division of United Aircraft, Hartford, Conn., and to Collins Radio Co., Cedar Rapids, Iowa. Each of these agencies provided comments of varying nature based upon either ducted propeller studies or related work.

The material compiled in this report forms a background of currently available information on ducted propellers. Future work in the field should be directed to fill the gaps and improve the basis for the design of these propulsion units.



ASSAULT TRANSPORT CONFIGURATION



Configuration

General

The four-duct tilting engine configuration powered by six Allison 550-B1 turboprop engines was selected as the best design to meet the requirements of the mission. The configuration which evolved during the course of the study employs four separate rotating ducted propeller units each with the power plants installed in the duct center body. A view of the over-all exterior configuration appears in Figure 29. An indication of the internal arrangement (Figure 8) is repeated here in enlarged form for reference purposes. The aircraft is characterized by a high wing, circular sectioned fuselage with integral rear loading ramp, and fuselage mounted landing gear. A three-man crew has been considered necessary to adequately perform the pilot and flight engineer functions for the aircraft.

In a VTOL aircraft the weight factor is all-important, so the utmost effort should be exerted to obtain a light, efficient structure taking advantage of advanced techniques and materials to achieve this end. Also, the equipment items should be selected carefully with due allowance for state of the art advances and restricting the systems to only those items which are necessary to perform the aircraft missions.

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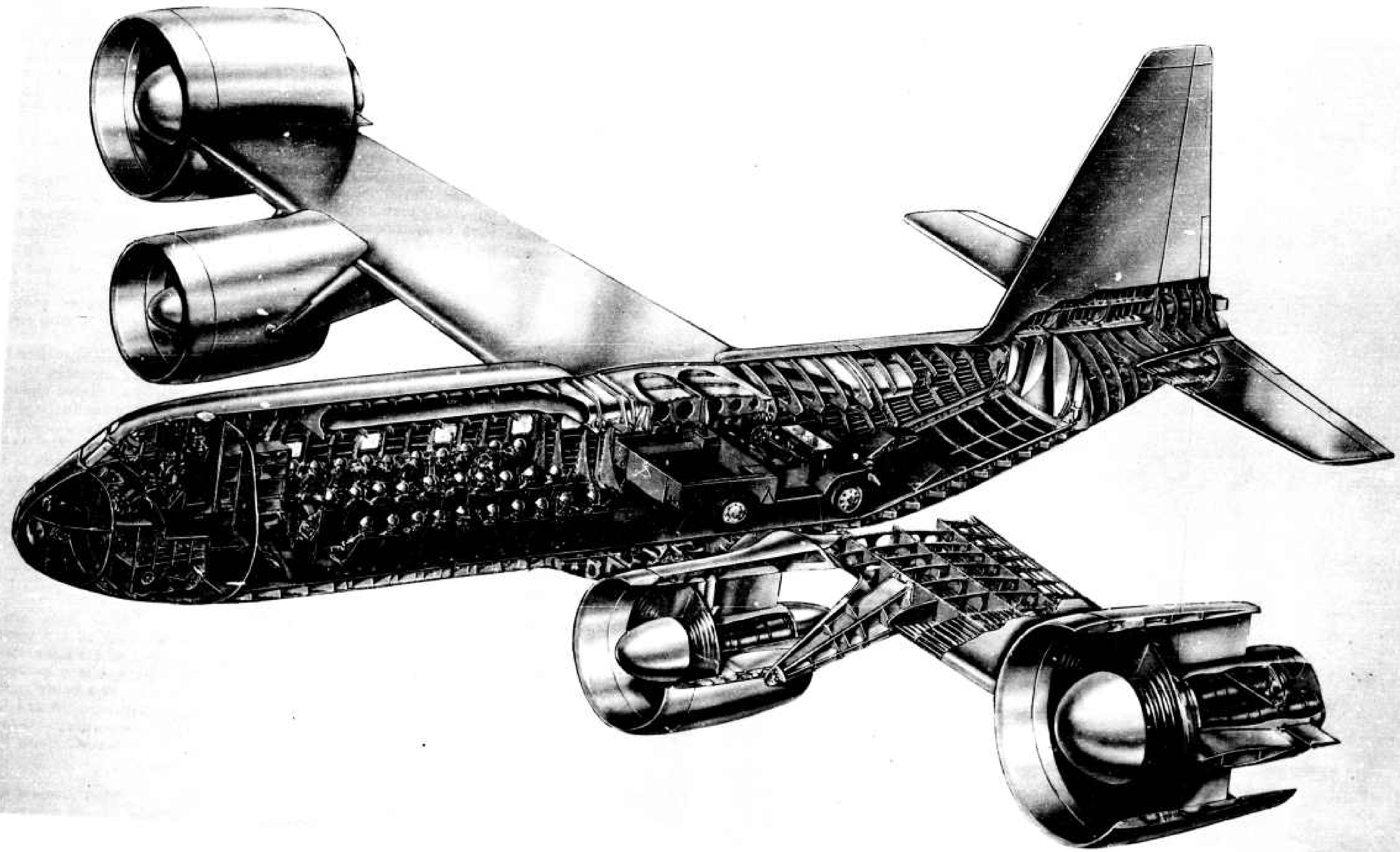

Figure 8. Cutaway of Assault Transport

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The Fairchild C-123 aircraft is the current operational Assault Transport and is in the same general weight class as the resulting ducted propeller transport designs. The C-123 was conceived, designed, and developed with the assault transport mission as its primary function. As a result, the same general approach was used in the determination of the design characteristics of the ducted propeller transports. Major emphasis has been placed upon the design of a minimum gross weight airplane to perform the required mission. Only the equipment which is considered necessary to perform the basic mission is included. However, space provisions have been allocated for other items which will be needed for alternate missions and aircraft loadings.

Description

Wing and Empennage

The main lifting surface is an unswept cantilevered high wing with an aspect ratio of about 5.8 and slight taper. The wing section selected was an NACA 64A412 cambered airfoil. A very simple and efficient three-spar wing structure was possible since there are no breaks in the primary structure and no flaps are required for landing or take-off operation. The supporting struts for the inboard ducts are cantilevered from the front wing spar and the supporting members for the outboard ducts are located at the tips of the wing structure. Analysis of the forces and moments exerted upon the wing indicate that the wing weight for this airplane will be less than the wing weight of conventional aircraft of similar size and wing loading. This is due in part to the increased relieving loads of the wing mounted ducted propeller propulsion units.

The tail surfaces are completely conventional in design, manufacture and function. No particular difficulty should be encountered in any phase of the development of these components.

Fuselage

The fuselage of the assault transport is constructed in the conventional ring, stringer, and skin fashion and should not require any new techniques or methods in its fabrication. The only unusual installation necessary for the VTOL operation is the location of the J85 turbojet engine in the extreme rear of the fuselage.

Ducted Propeller Units

The ducts are constructed in a manner similar to the fabrication of turbojet pod nacelles. Maximum stiffness at the rotor station will be maintained to eliminate rubbing of the rotor blade tips against the shroud inner surface. The centerbody is held rigidly in the duct by means of struts in the aft centerbody area. The powerplants are mounted within the duct centerbody shell using the normal engine mounting pads.

In this transport configuration, the propellers are mounted directly on the engine reduction gear output shafts. The reduction gear boxes are considered to be very similar in design and construction to existing engine-mounted reduction gears. For the outboard units in which the engines are coupled, the existing system of engine clutches and gear trains to the coaxial output shafts can be used in the design of the propeller drive system. It is evident, then, that the reduction gear box development problem on the ducted propeller units will be comparable to those encountered on present-day turboprop engines.

It is quite practical to consider the development of the complete ducted propeller unit in a single integrated program. One of the existing industry powerplant-propeller teams would be admirably suited to undertake the design, development and fabrication of the complete propulsion package so the airframe manufacturer would simply install the units on the airplane.

This concept of packaged units is attractive in the consideration of maintenance since the total assembly can be easily replaced and would be serviced as a single powerplant.

In this scheme, the introduction of advanced or modified engines could be easily accommodated by construction of revised ducted propeller units without the necessity for any major airframe changes.

Landing Gear

A tricycle type landing gear is provided. The nose gear has dual wheels and is conventional in design. Retraction of the nose gear is forward into the fuselage and is accomplished electrically. Hydraulic nose-wheel steering and parking brakes are included. The main gear has two tandem wheels located on each side of the fuselage. Retraction of this gear is inboard into the fuselage and is accomplished electrically. Position indicator and controls and warning lights are located in the crew compartment.

Inboard Profile

The inboard profile of the transport fuselage was the object of considerable design effort and is shown in Figure 8. The cockpit arrangement for the three-man crew was the subject of considerable design study. A circular cross-section was used for the pressurized body, and the cargo and troop accommodation problems were studied in quite some detail. Troop seat and litter arrangements were varied to find optimum loading of the cargo compartment. Alternate cargo loadings were also considered. The large cargo loading ramp and door arrangement was studied.

In connection with the work on the inboard profile, research into the various items of fixed equipment was initiated. Data were collected and examined on instruments, crew furnishings, controls, air conditioning and pressurization equipment, electrical and communicating equipment, and survival gear. In addition, auxiliary power supply and cargo handling and tie-down equipment were considered briefly. The result of the inboard profile study is presented in Figure 74. As stated previously, not all of the equipment described is installed in the basic aircraft but provisions are provided for installation when required.

Crew Compartment

The crew, consisting of the pilot, co-pilot and systems engineer is provided with a pressurized and environment-controlled compartment. The control arrangement is standard, and the location of the instruments and windows are such as to provide excellent visibility for the pilot and co-pilot. The systems engineer, located aft of and between the pilots, is provided with an adjustable swivel seat to allow him easy access to the controls and instruments on the center console, as well as the switches and instruments on the panel located behind the co-pilot. Entrance to crew compartment is through a large door on the left hand side of airplane and up a stairway. An emergency exit is provided in the top of the fuselage just aft of crew compartment or through the entrance door which is jettisonable.

Cargo Compartment

Due to the fact that the fuselage is fully pressurized, a circular cross-section was maintained as nearly as possible. Therefore the width of the cargo compartment is greater at the center than at the floor level. The volume of the compartment is approximately 2600 cubic feet and is capable of taking a maximum varied cargo among which are the following:

59 Assault or paratroops

52 Litters

1 Field Ambulance 1-1/2 ton 4 x 2

2 Field Ambulances 3/4 ton 4 x 4

2 Cargo Carriers M29C (Amphibian)

1 Scout Car M3A1 4 x 4

2 Shovel Loaders, Tractor Mounted, Hydraulic, 1/3 cubic yard

1 Tractor, High Speed 7 ton, M2
Various Small Trailers

3 Jeeps

2 Trucks 1/2 ton 4 x 4

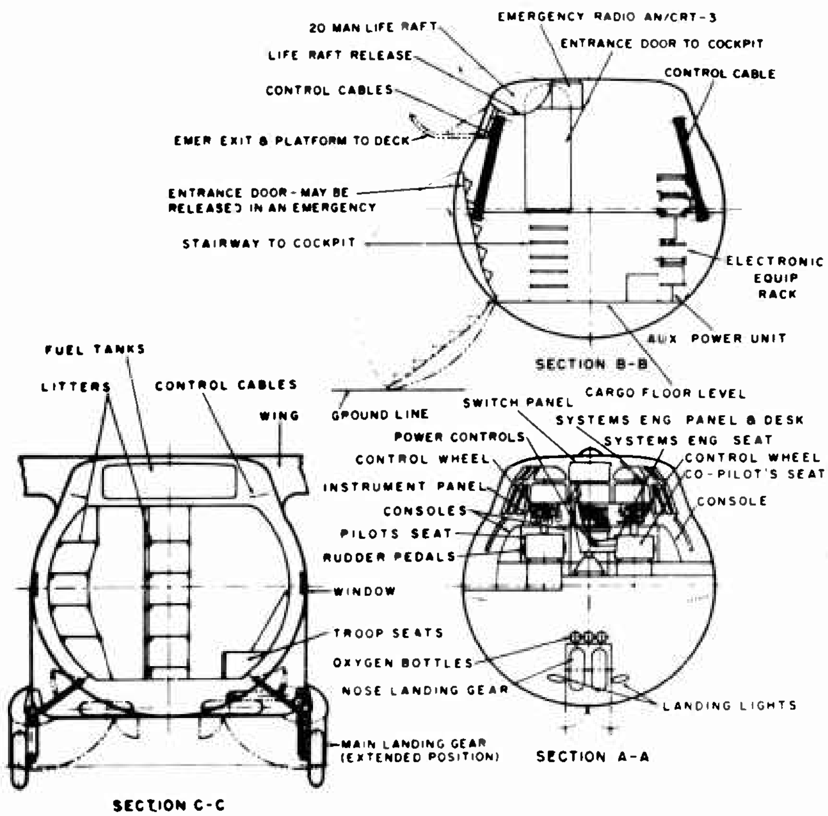


Figure 74. Inboard Profile (Sheet 1 of 2)

2 Trucks 3/4 ton 4 x 4 Weapon Carrier

105 mm. Howitzer and Truck 3/4 ton 4 x 4

Representative VTOL and STO payloads are presented in Table II.

Large doors are located at the aft end of the compartment for rear loading. The floor level is 48 inches above the ground level (average truck bed level) and is capable of withstanding loads imposed by the above equipment. The doors are pneumatically operated (the air source being from the J85 engine to accumulators) and the ramp door can be raised to intermediate levels for convenient truck bed or platform loading as well as ground level ramp loading. Cargo, troop seat, and litter hold-down fittings are provided in the floor and are located according to HIAD AD7. Fuselage frames and longitudinal beams are so placed that each fitting is located at the intersection and is properly reinforced under the floor. Troops may enter through the cargo ramp door. This ramp may be used for bail-out of paratroops or dropping of cargo. Doors are also provided on each side of the fuselage adjacent to cargo doors for evacuation of troops. In case of ditching, emergency exits are provided in the top of fuselage, also in the sides just forward of the wing.

Survival Equipment

Two automatically inflated 20-man life rafts are provided. These are in built-in containers in the top of fuselage. The forward one is located aft of the crew compartment, access to which is through an emergency door and platform, located on the left side. The rear one is located aft of the cargo compartment, and adjacent to an emergency exit. A ladder is provided to this exit. Additional manually operated life rafts will be lashed near appropriate exits when carrying a full complement of troops. An emergency radio is located adjacent to the forward life raft and is accessible through the raft compartment when the raft is released.

First aid kits are located in the crew compartment and also in the cargo compartment.

Aircraft Systems

Aircraft Controls

The aerodynamic surfaces are actuated by dual cable systems. Balance of the elevator and rudder surfaces, and sealing between the fixed and moveable surfaces, reduces the hinge moments and allows the use of a mechanical control system without the aid of power assists. Cables are routed so as to keep the bends at a minimum, thereby reducing friction.

Trim tabs are provided on the rudder and elevators and the electric actuators are controlled by switches located in the crew compartment.

Longitudinal control during hovering or low speeds is provided by the jet thrust from the J85 engine located in the aft end of the fuselage. Lateral and directional control is provided by controllable surfaces located in the aft portion of the outboard ducts.

Powerplant and Controls

Six Allison 550-B1 engines are installed in ducted nacelles in the wing. Each inboard nacelle contains a single engine and propeller, while the outboard nacelles contain two engines driving contrarotating propellers. An automatic torque sensing device is incorporated for equalizing opposing engines, that is, if one engine loses power or fails, the engine on the opposite side is cut back or shut off. The power controls and engine switches are located on the console between the pilots. Engine instruments are installed on flight engineer's panel.

For VTOL or STOL operation, the nacelles (ducts) are rotated by electric actuators. Switches on the control wheel of the pilot and co-pilot initiate the rotation and indicators on the instrument panel show the extent of rotation. For VTOL the nacelles are rotated 90°, and for STO (short takeoff), approximately 45° aft. When decelerating for vertical landing, the nacelles are rotated 10° forward of vertical. The actuator motors are powered by the primary electrical system. However, a separate

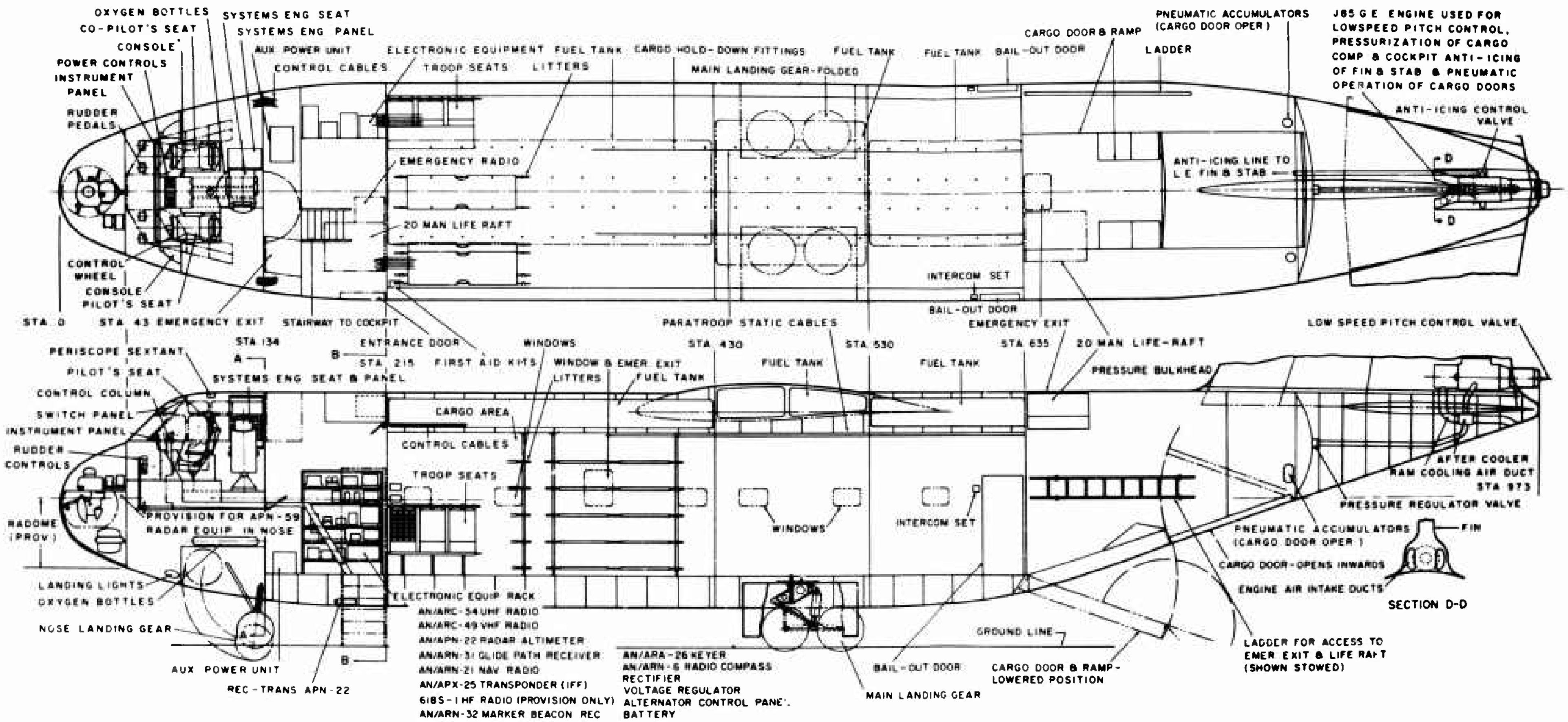


Figure 74. Inboard Profile (Sheet 2 of 2)

emergency motor, powered by the auxiliary power system, is provided in the event of failure of the primary system. This emergency motor drives a coordinating shaft which rotates all the nacelles.

Environmental Control System

The crew compartment and the entire fuselage, with the exception of the loading gear wheel wells, back to the pressure bulkhead, aft of the cargo doors, are pressurized, heated and cooled. The source is a bleed from the J85 engine located in the aft end of the fuselage. This same source is utilized for anti-icing of the stabilizer and fin. A combustion heater is used for anti-icing the wing and supplements the engine supply for heating and defrosting the crew compartment. The necessary pressure regulators, heat exchangers, etc. to maintain the environment at a level for efficient operation are provided. A differential pressure to maintain a cabin altitude of 8000 feet at the normal flight altitude will be supplied.

Navigation System

The navigation system includes the following equipment:

AN/APN-22	Radar Altimeter
AN/ARN-31	Glide Path Receiver
AN/ARN-21	Radio
AN/APX-25	Transponder
AN/ARN-32	Marker Beacon Receiver
AN/ARN-6	Radio Compass

Provision is made in the nose of the fuselage for the installation of AN/APN-59 radar equipment. This set has search, weather mapping and terrain mapping capabilities.

The majority of the above equipment is located in racks in the section just aft of the crew compartment and is easily accessible for servicing and adjustment. The indicators are located on the instrument panel and the control boxes on the console between the pilots.

A periscopic sextant is provided in the crew compartment for celestial navigation.

Communication System

The communications system includes the following equipment:

AN/ARC-34	UHF Radio
AN/ARC-49	VHF Radio

Provisions for the installation of 618S-1 HF Radio

AN/ARA-26	- Keyer
AN/AIC-10	Intercommunication set with two stations in cargo compartment and one in crew compartment.

An emergency radio set CRT-3 is located in top of fuselage and is accessible through the life raft door. The above listed communication and navigation equipment are the most advanced items with available data. However, full advantage will be taken of equipment modernization whenever it is possible to do so.

Electrical System

The primary electrical power source is from two alternator mounted on the accessory pads of the inboard engines. This a-c system is fully automatic in operation and incorporates a maximum of automatic circuit protection features. Direct current of 28 volts is obtained by rectification of the primary a-c current.

Emergency power, in the event of failure of the primary source, is provided by a gas turbine auxiliary power unit furnishing a-c power.

A-C and d-c external power receptacles are provided for ground operation and testing.

Oxygen System

In the event of pressurization failure, a low-pressure oxygen system is provided for the crew only. Masks, demand regulators, and pressure gages are located at each crew station. A filler valve is located in the nose wheel well.

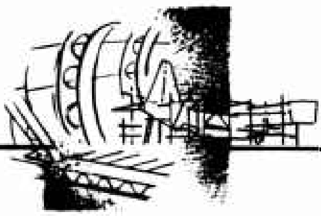
Hydraulic System

A 3000 psi hydraulic system is provided as the power source for nose wheel steering and parking brakes. Engine-driven hydraulic pumps are installed on each of the two inboard engines.

Fuel System

The fuel system consists of 6 fuel tanks pressurized to 5 psi by engine bleed air. Two tanks are located in the fuselage, one forward and one aft of the wing. The other four tanks are located in the wing center section. The total capacity is 13,295 pounds, (2045 gallons). An additional 12,700 pounds of fuel may be carried in the wing. Fuel is programmed to keep cg movement to a minimum, and the tanks are so interconnected as to make fuel available to any engine without interruption of flow. When required, fuel cells can also be located in the cargo compartment.

The system feeds the G.E. J85 engine located in the aft end of the fuselage as well as the six Allison 550-B1 engines in the wing nacelles.



Structures

General

The VTOL Ducted Propeller Assault Transport, configuration D181-960-009, was analyzed and described from a structural standpoint. Structurally the transport is conventional in both arrangement and construction with the exception of the rotating ducts and engines. The location of these ducts and engines in the wing, influences wing structural design because of the large mass of the units and also because of the high thrust values inherent in the VTOL aircraft. The fuselage is constructed similarly to other transport types, and has large cut-out areas for cargo loading and landing gear.

Appropriate additions or modifications have been made to the current conventional aircraft conditions in MIL-A-8629 (Aer) in order to provide structural criteria for ducted propeller aircraft. The gust condition is found to be the most critical flight condition; however, the wing structure is primarily designed by the vertical take-off and taxiing conditions. Preliminary loads were estimated for the various loading conditions to permit estimation of required structural sizes and materials.

Design Criteria

The applicable design specification is MIL-A-8629(Aer). The assault transport under investigation is classed as type VR and is designed for a maximum symmetrical flight-maneuver load factor of 3.0. However, the airplane strength level for flight is partially determined by gust considerations which require a design limit load factor of 3.2. The wing basic structure is primarily designed by the vertical take-off condition at a vertical load factor of 1.75. Because of the large concentrated weights of the ducts and engines, the wing lower surface may be designed by the ground taxiing condition requiring a load factor of 1.67. The airplane is designed for vertical landing only, at a load factor of 2.67. Since a maximum oleo deflection of 10 inches is provided, an estimated oleo efficiency of 75 percent is required. Since the engines (ducts) are designed to be rotated in an approach to vertical landing, the design conditions applicable to the conventional landing-approach will be applied to the transition configuration (when the engines are rotated from the normal horizontal attitude). Therefore, for structural design purposes, the landing approach limit speed is selected as the maximum design speed for the transition. This criterion seems reasonable since it is anticipated that the rotation of the engines will commence at about 120 knots in the typical approach for a vertical landing. The landing-approach limit speed is 175 knots.

In load calculations involving the transition, it must be considered that the engine thrust vertical component and the aerodynamic lift on the rotated ducts will be providing additional components to the lift given by the conventional surfaces.

All control systems are unassisted by power boost mechanisms. However, longitudinal control during hovering or at low speeds is provided by the jet thrust from a J85 engine located in the airplane aft fuselage in the region of the vertical tail. Lateral and directional controls are provided by controllable surfaces located in the aft portion of the outboard ducts. These surfaces when deflected are loaded by pressures resulting from the ducted propeller exit velocities.

The unique design of an aircraft with rotating engines necessitates consideration of certain design conditions not covered by the applicable specification.

The estimated weights used for structural analysis are presented in Table IX.

Table IX STRUCTURAL GROSS WEIGHTS

CONDITION	WEIGHT pounds	cg Percent MAC
Empty	42,000	17.0
Minimum Flying	45,000	----
Basic Landing	58,000	----
Basic Flight	61,000	----
Basic Take-off	68,000	17.6

Structural Description

Structurally the transport is conventional in both arrangement and construction with the exception of the rotating ducts and engines. Aluminum alloy, stringer-stiffened shell structure is used for the pressurized fuselage and the lifting surfaces. The fuselage structure contains a number of door and window cutouts, typical of a transport; in particular there is a large cargo loading door in the rear lower surface of the fuselage. All cutouts are longeron reinforced. The location of the ducts and engines in the wing, influences wing structural design because of the large mass of the units, and also because of the high thrust values inherent in the VTOL aircraft. Each ducted fan, complete with engine, is carried in a nacelle structure, supported in turn by radial spokes within the ducts. The complete duct assemblies, one at each wing tip and one at the 60 percent of span station of each wing panel, are hinged about the pitch axis. The basic structural arrangement can be seen in Figure 8.

Lifting Surfaces

The lifting surfaces (wing, horizontal and vertical tails) are stringer stiffened covers of 7075-T6 aluminum alloy material with three spanwise shear webs. This type construction is the optimum structure for the low intensity cover loading present in this configuration.

The wing is made up of two panels with the structural section, between the front and rear spars, carried through the fuselage. These two panels are spliced together at the airplane centerline by means of match angle fittings. Ribs are provided at the splice to distribute the loads. The wing-fuselage attachment is accomplished by bolted connections at four points. Fittings are provided to distribute the loads to the front and rear spars and a root rib. Fittings and ribs are also provided at the inboard and outboard duct support points to distribute the loads from these ducts into the wing structure. Because of the large masses of the ducts located outboard on the wing, the wing has been designed for compression in both the upper and lower surfaces. Critical conditions are vertical take-off (compression in the upper surface) and taxiing (compression in the lower surface). Ribs have been spaced at 20 inch centers, along the wing span to stabilize the stringers.

The vertical tail is attached to the fuselage by six bolts through fittings which are attached to the three spars and a closure rib. The load is distributed to the fuselage by fittings fastened to three fuselage frames. The horizontal tail is fabricated as two outer panels, fastened directly to the fuselage by match angle fittings. The mating fuselage frames provide the stabilizer carry-through structure across the fuselage and supply the bending rigidity required. This is accomplished by providing a web with upper and lower caps across each frame. Large doublers at the stabilizer root, collect the stringer loads and concentrate them at the spar caps. Ribs have been spaced along the span of both the fin and stabilizer, to stabilize the stringers and also to distribute the concentrated hinge loads from the rudder and elevator.

Fuselage

The fuselage is constructed primarily of stringer stiffened skin in 2024-T3 aluminum alloy, stabilized by frames. This construction is again dictated by the low axial loading in the skin, resulting from the large depth and breadth of the fuselage. Since the fuselage is pressurized, but is not completely circular at all stations, the stringers are also required to carry pressure loads not resisted by skin tension.

The fuselage contains a number of doors and windows, a cutout for the wing, and a large cargo loading door in the lower surface at the rear. Reinforcements around these cutouts are sufficiently extensive so that four continuous longerons are required. Heavy frames are provided to distribute wing, tail surface and landing gear loads.

The cabin area is designed to maintain 8000 feet pressure altitude at 30,000 feet actual altitude and a domed bulkhead is provided at the rear to terminate the pressurized area. Since the cargo loading door removes a large area of the lower fuselage shell, provision is made in the door fastenings to carry the "bursting" loads due to pressure.

The lower part of each frame in the cargo compartment area forms a deep cross-beam supporting the cargo floor, constructed of aluminum sheet stiffened by longitudinal angle section stringers. Fuel is carried in a flat lined cell between the cargo compartment ceiling and the top outer skin. The cargo compartment ceiling is stringer stiffened to carry the fuel weight, with the area around the cell vented to cabin pressure. This results in the pressure load being carried by the outside fuselage shell.

Ducts

The inboard duct contains an Allison 550-B1 engine which is mounted conventionally to two rings in the nacelle-type structure forming the hub. The hub is attached to the duct shroud by a set of four spokes, two of which contain trunions about which the entire duct rotates. These spokes extend some distance along the ducts to provide fairing and fore and aft stiffness. An actuator is provided to rotate the ducts and to maintain them statically during normal operation. The annulus type duct has an airfoil cross-section, and consists of an inner and outer skin separated by ribs. The circular shape is maintained by rings, one of which is placed in the plane of the fan to maintain a small gap between the inner wall of the duct and the fan. This latter is a requirement for high fan efficiency. Two truss type struts, with skin fairing, support the duct and distribute the loads to the front and center wing spars and supporting ribs. The critical design condition for these struts is a lateral load from the duct assembly.

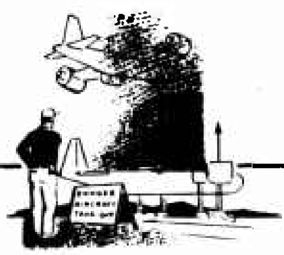
The outboard ducts are similar to the inboard ducts except that there are two Allison 550-B1 engines mounted in the hub, and the whole duct assembly is supported by a 10 inch steel tube cantilevered from the wing tip. This tube transmits the loads, through bearings and fittings, directly into the wing structure. It also serves as a torque tube providing the duct rotation. The critical condition for this tube is vertical take-off which imposes the maximum bending moments.

Landing Gear

For landing, a tricycle type gear mounted in the fuselage has been provided. The nose gear has a dual wheel and is conventional in design. The main gear consists of two tandem wheeled gears mounted at each side of the fuselage. Each gear is mounted to a single fitting hinged to the fuselage, thereby making it possible to fold the gear into the fuselage. Because of this, the side load on the gear, which imposes torque on this fitting, is the critical design condition.

Weight and Balance

The estimated weight of the Assault Transport, reference BAC Drawing D181-960-009, is presented in Table VII. Conventional methods of weight estimation are used in determining these structural weights. Duct weights are based on data available from previous Bell Aircraft Corp. duct designs.



Aircraft Safety

General

The problem of safety is encountered in aircraft of every type. An additional factor which must be considered by the designer of VTOL aircraft is the loss of power in the hovering and transition phases of the flight plan. The study requirement for the aircraft stipulates that the aircraft should be capable of a controlled crash landing in case of emergencies arising during hovering flight.

It is generally accepted that the VTOL aircraft type under consideration will not be designed with hovering as its principal mission. The conception has been that the aircraft will pass through the vertical and transition flight phases as quickly as practicable during landing and take-off operations. Missions requiring any lengthy hovering time are considered secondary to the primary use as assault transports.

The safety and survival of the aircraft, crew, and passengers is the principle objective of any investigation of the problem. Therefore, the study was undertaken to provide some insight into the major design parameters which must be considered in the successful solution of the hovering and transition safety problem. The investigation may be subdivided into several definite phases, each amenable to study as a problem in itself. The attempt here has been to delineate these lines of study and to determine the quantitative range of values for the factors involved. The investigation is roughly organized into three areas: (1) Aircraft Behavior, (2) Physiological Effects, and (3) Design.

Aircraft Behavior (Hovering Failure)

Acceleration

The complete or partial failure of the vertical thrust producing units on the hovering aircraft will result in a net accelerating force in the downward direction. The magnitude of the accelerating force is directly proportional to the extent of power failure. In the multiple engine aircraft which are presently under consideration, the possibility of total failure or 100 percent thrust loss is extremely remote. Consider the case of a six engine configuration which was presented in Figure 5. Curves of impact velocity variation with height of fall are shown for the case of four, two or zero engines operating. The maximum hovering altitude has been shown at 50 feet, the obstacle height requirement for performance of these aircraft. With four engines operating, the impact velocity from 50 feet is about 21 mph and time of descent is about three seconds. At the extreme case of full power failure, the impact velocity is 38 mph in about 1.8 seconds time. These values have been computed with the assumption of instantaneous power loss which is not representative of the probable actual conditions. The inertial-energy of the rotating components would result in a gradually reducing thrust level.

Deceleration

The impact of the descending aircraft with the ground will result in the deceleration of the aircraft and its contents. The exact nature of the deceleration process is very complex and not easily amenable of solution. However, a very general type of study can be made to determine the magnitude of the average loading which must be applied to the aircraft in order to come to a complete stop. The result of this study is presented in Figure 75. Here

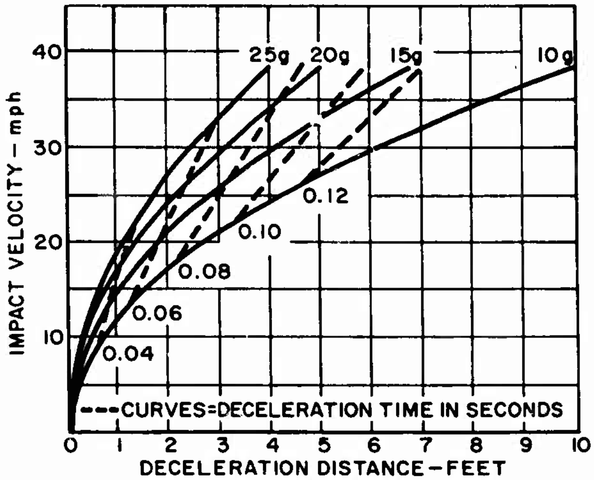


Figure 75. Deceleration Time

the variation of deceleration distances required is shown versus impact velocity. A family of such curves is presented to show the effect of different constant deceleration g loadings. The time required for constant deceleration at the different loadings can be read from the plot. For instance, a 15 g deceleration with an initial impact velocity of 25 mph would be accomplished in about three feet of distance and require about 0.88 seconds of time.

Physiological Effects

As the aircraft decelerates, the crew and passengers will also be subject to deceleration. In each case the rate of deceleration on set and the peak and average forces which act upon them cannot be determined without exhaustive investigation and analysis. However, the general information on the limits of human tolerance to the type of g loading which can be expected during an emergency landing of the VTOL airplane can be presented (Figure 26). Examination of this data (Reference 19) shows that even under the extremely pessimistic assumptions for acceleration and deceleration presented in the preceding figures, the human resistance to the decelerations imposed for the periods of time required are still within the limits of human tolerance. The example ex-

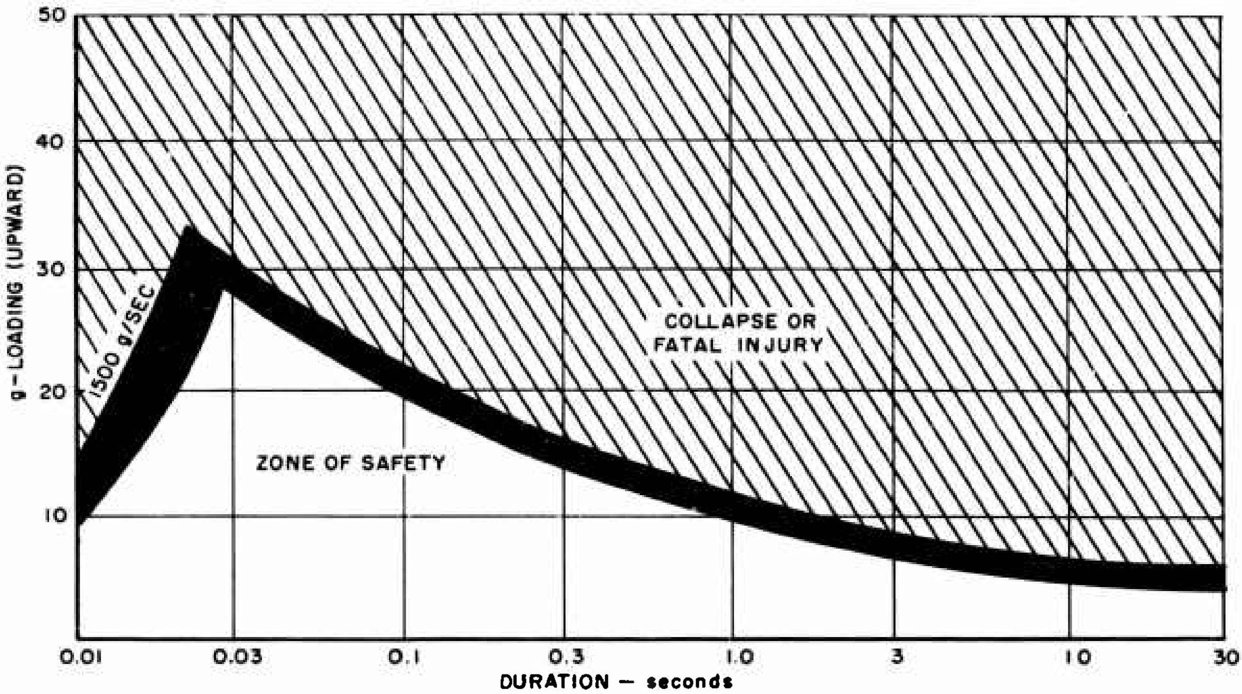


Figure 76. Limits of Human Tolerance for Escape Systems

amined previously in which a 15g deceleration is imposed for 0.08 seconds can be seen to be well within the tolerance boundary. This assumes that the person is rigidly attached to the structure and is subject to the full deceleration experienced by it. In actual conditions, this is seldom the case.

Design for Safety

Aside from the universal concept of ultimate system reliability, there are many other features which can be incorporated in the design of an operational assault transport to minimize damage and injury in case of emergency situation.

Aircraft Control

A stability and control system must be designed for the aircraft for operation in the vertical take-off and landing phases of the flight plan. In the event of a failure during these operations, sufficient control must be retained to successfully maintain the aircraft attitude down to the ground. This point has been considered to be of primary importance in the determination of the stability and control system. An automatic thrust equalization device is considered a necessary part of the system. This would prevent uncontrollable roll forces which might result from a propulsion system failure on one side and would allow adequate control in the emergency.

Aircraft Structure

The total kinetic energy of the falling airplane must be absorbed at a rate consistent with the safe deceleration values established for the design. This energy must be dissipated by components of the aircraft structure at the required rate. This presents a formidable problem to the designer of the aircraft in that it would be desirable to c tain this characteristic without undue penalty in structural weight.

It is expected that the extended landing gear and the lower part of the fuselage structure will be used as the energy absorbing components. How this will be accomplished was beyond the scope of the present study, but should be the subject of a detailed investigation as soon as possible. The landing gear can be considered to absorb a considerable amount of the energy since it is expressly designed to perform this function. Normally the gear is designed for specific values of sink speed and deceleration desired. Vertical rates of 8 to 12 feet per second are normal in gear design. It can be expected that the gear can be designed to absorb the same amount of energy in case of an emergency descent. The kinetic energy remaining must be dissipated by deformation, buckling and failure of the tires, landing gear structure and the lower fuselage structure. At this time it is not known whether any radial design changes must be made to do this. Another approach might be the use of special lightweight energy absorbing material placed in the lower fuselage compartments. A considerable amount of energy may also be absorbed by deformation and displacement of the surface upon which the aircraft falls.

Personnel Protection

Simple design features may also be incorporated into the personnel furnishings for the aircraft which will decrease the decelerations to which the crew and passengers will be subjected in case of emergency. The present troop seat construction using web and fabric appears quite good in that the material can stretch under the loads imposed. This allows restrained motion of the passenger downward during deceleration thereby relieving the peak loads which are imposed. In addition, the use of seat structure which will yield under loads greater than the dangerous tolerance levels, could serve to maintain the g loadings at a safe value for the occupants. This would also be an excellent field for additional design development.

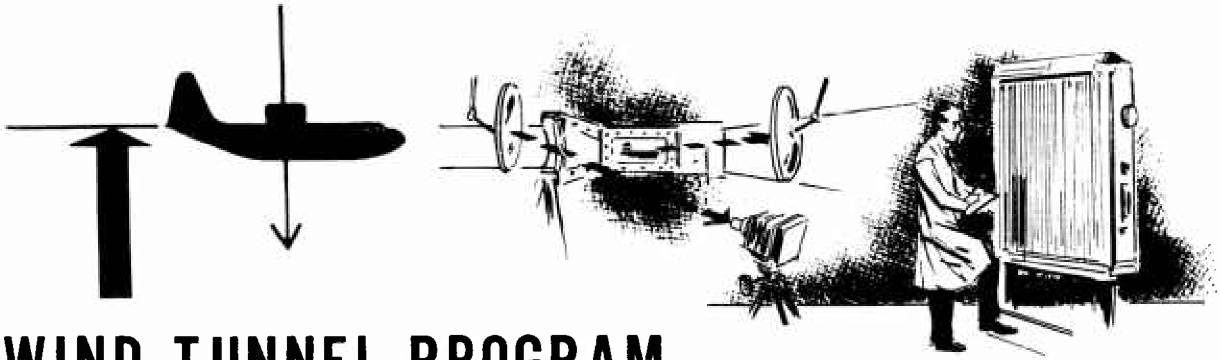
To summarize, it may be stated that the basic problem of emergency landing arising from power loss during the hovering regime appears soluble. Under the extreme conditions assumed for this analysis, a solution to the problem of crew and passenger safety appears achievable although much basic investigation remains for a satisfactory analysis and solution.

Transition Failures

The determination of total time required to perform the vertical take-off and landing sequences were discussed in the Section V, Aerodynamics. The study of power required during the various phases of the transition was also discussed. It was shown that 12 seconds after start of transition, forward flight can be sustained with the power available from only two of the six engines. This time-velocity-power data is shown in Figure 16. The comparison of the power required for other types of VTOL aircraft of interest is also presented.

It can be seen, that the time period in which a single engine failure can be dangerous is very small when the aircraft is in the transition phase and that this characteristic is similar for the various types of which are presently considered. In this danger zone the aircraft will still be controllable in event of failure and the situation would be comparable to that of the hovering case without any more serious consequences expected upon impact.

In forward flight the aircraft will be safer than current transport aircraft since the power installed for VTOL performance is in the order of three times that required to sustain normal level flight. In event of emergency under these conditions the aircraft can operate as a conventional or STO configuration which is possible with the wheeled landing gear.



WIND TUNNEL PROGRAM

Program

Early in the study, arrangements were made to conduct a ducted propeller model wind tunnel program, at the facilities of the University of Wichita. The program called for a redirection of the work projected for the University under its current ONR contract. Two model ducted propeller units were established for test by the Bell Aerodynamics Section. The design, fabrication, and test of the models was undertaken by the University staff.

Analysis and Design

The basic aerodynamic characteristics of the models to be tested by the University of Wichita were determined by the Bell Aerodynamics Section in its exhaustive studies of the ducted propeller units. The models which were defined represent 1/10 scale models of the 200 knot T49 configuration shown in Figure 77 and the RB109 variable pitch configuration in Figure 78. A slight modification has been made to the T49 design, in that the number of inlet stators has been changed from 21 to 18. Model specifications are included as part of Reference 2.

The design work performed by Bell Aircraft personnel was in the nature of technical coordination between the University of Wichita and Bell Aircraft. Basic model layouts were pre-

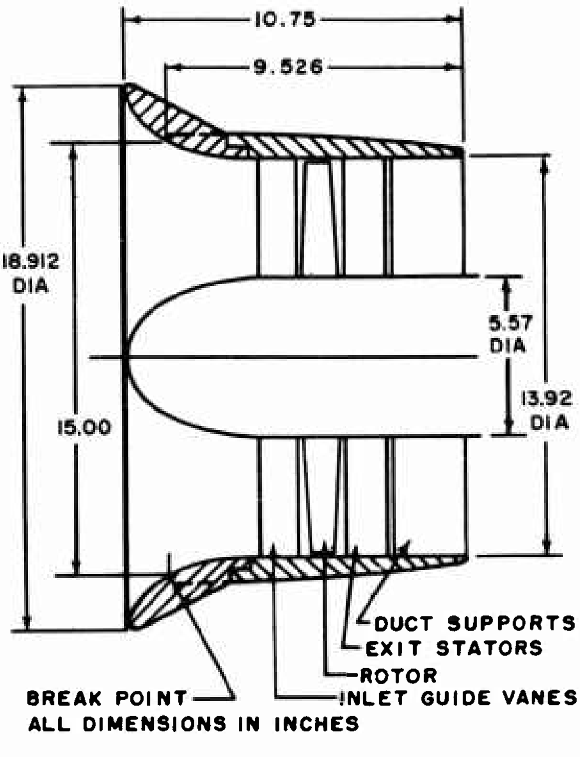


Figure 77. Wind Tunnel Model No. 1. Fixed Rotor Blade and Stators with Adjustable Inlet Guide Vanes

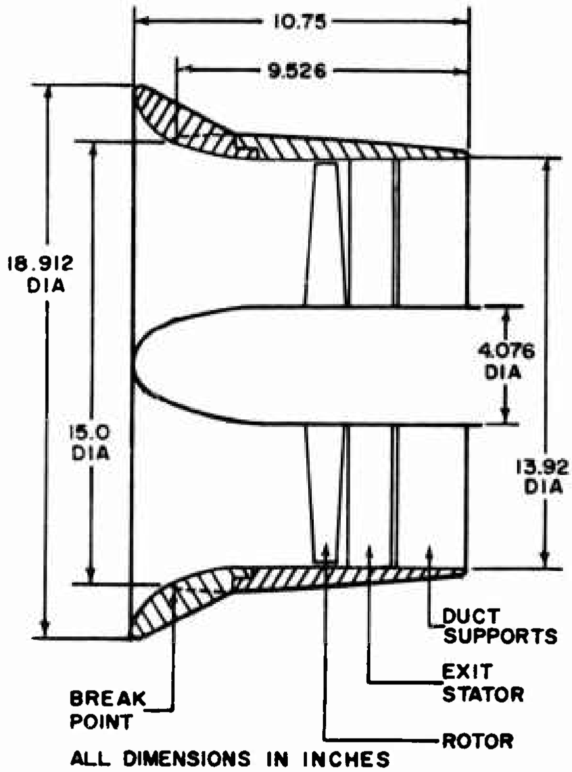


Figure 78. Wind Tunnel Model No. 2. Variable Pitch Rotor and Fixed and Fixed Exit Stators

pared to transmit the necessary dimensional data to the University for the detail design of the test equipment. This was done for both of the test models selected. This section coordinates of the inlet vanes, rotor blades, and exit stators had been determined by aerodynamic analysis. This information was converted to very accurate drawings by the Bell loft department to a large scale. These sections were then reduced photographically to full size and metal templates produced from the negatives. The metal templates were used by the University in the manufacture of the model blades.

Stress analysis of the critical structural items of the models were performed by the Bell Aircraft Structures Section and the results used to select the materials needed for the various components of the models.

Testing

On April 24, 1956, the wind tunnel tests were started on a model of the T49 configuration re-

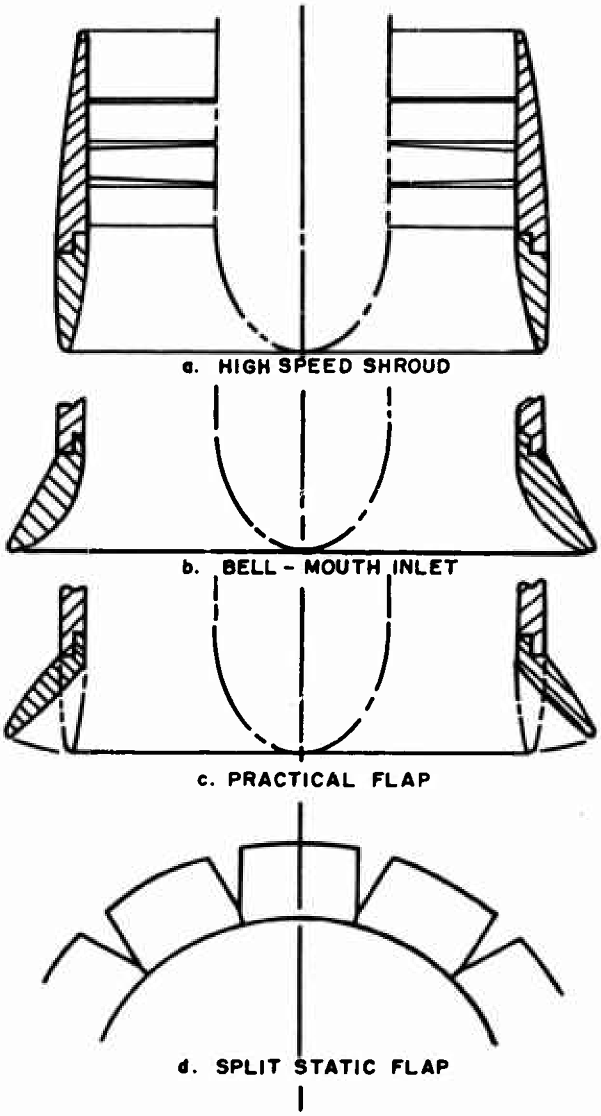


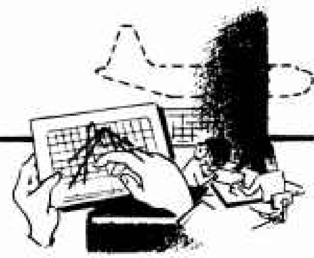
Figure 79. Wind Tunnel Model Duct Inlet Variations

sulting from the theoretical methods developed in the Bell Aircraft study. Both force tests and pressure tests will be conducted during this program. Angles of yaw from 0° to 90° will be examined to evaluate hovering and transition conditions. The inlet sections of ~~the~~ models are interchangeable as shown in Figure 79, so that both the low drag high speed and flapped high thrust static inlet configurations may be tested. The rotor for the fixed bladed design model No. 1, is shown being balanced in Figure 12. Figure 13 shows a partial assembly which reveals the 75 horsepower electric motor, the shafting and the rotor sting mounted in the shop. Figure 14 shows the model No. 1 mounted in the University of Wichita 7 foot by 10 foot tunnel. The high speed inlet is in place. The results of these tests will be published by the University of Wichita in a future report.

Bell Aircraft Aerodynamics personnel were present at the University and assisted in the testing and analysis of the data. Automatic recording oscillograph equipment was loaned to the University by the Bell Aircraft Instrumentation Laboratory. An instrumentation engineer assisted in the installation and calibration of this equipment prior to the start of the tunnel test period.

The initial testing period has been completed and the very preliminary data indicate that the results of the testing agree quite well with the original analysis. The model was operated at full design power and was yawed to the tunnel airstream at angles to 90 degrees. Tests were performed with the various duct inlet leading edge configurations and it appears that the modified bell mouth practical flap inlet configuration will be very satisfactory for the static take-off condition.

CONCLUSIONS AND RECOMENDATIONS



Conclusions

As a result of this study, the following conclusions have been reached:

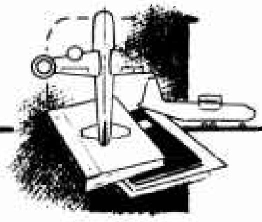
1. A very practical VTOL Assault Transport Aircraft can be designed with a ducted propeller propulsion system. The resulting configuration can be simply constructed and is capable of exceptional high-speed performance and good cruise economy.

2. The operation of the aircraft in the STO (short take-off) mode greatly increases the pay load or range capability of the basic VTOL design. The operational flexibility and versatility of the aircraft, with this STO/VTOL characteristic, enables it to be utilized for a wide variety of transport missions.

3. The use of rotable ducted propeller units with integral turboprop engines, located in the duct centerbodies, appears to result in the most suitable aircraft configuration from the standpoints of weight, simplicity of construction, development, and maintenance.

4. The VTOL aircraft which were studied, can be controlled during emergency descents resulting from power failures in the VTOL phases of operation, thus assuring the safety of the occupants and minimizing damage to the aircraft.

5. The study was based upon components which will be available by 1960 and indicates that the aircraft can be operational in the 1960 time period.



Recomendations

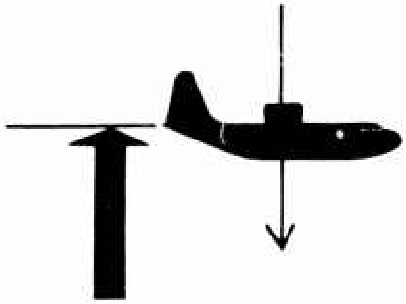
The general recommendations which can be made from the knowledge gained during this study are summarized here.

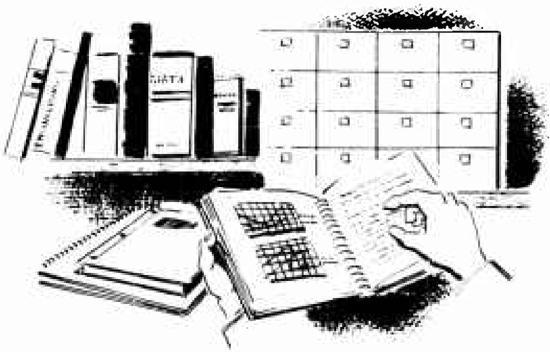
1. It is recommended that theoretical and experimental investigations of basic and applied ducted propeller characteristics be continued and expanded.

2. It is recommended that turboprop development programs be implemented to include (a) water injection systems, to maintain sea level rated power at extreme altitude and high ambient temperature take-off conditions; and (b) lubrication and bearing system, to enable the engines to operate at all angles of tilt up to 110 degrees from the normal horizontal position.

3. It is recommended that a program be initiated immediately on a research or test-bed airplane embodying the basic principles of the ducted propeller propulsion concept studied at Bell Aircraft.

4. It is recommended that a program leading to the accelerated development of a VTOL ducted propeller transport aircraft, based upon the results of this study, should be started without delay to achieve operational availability at the earliest possible date.





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